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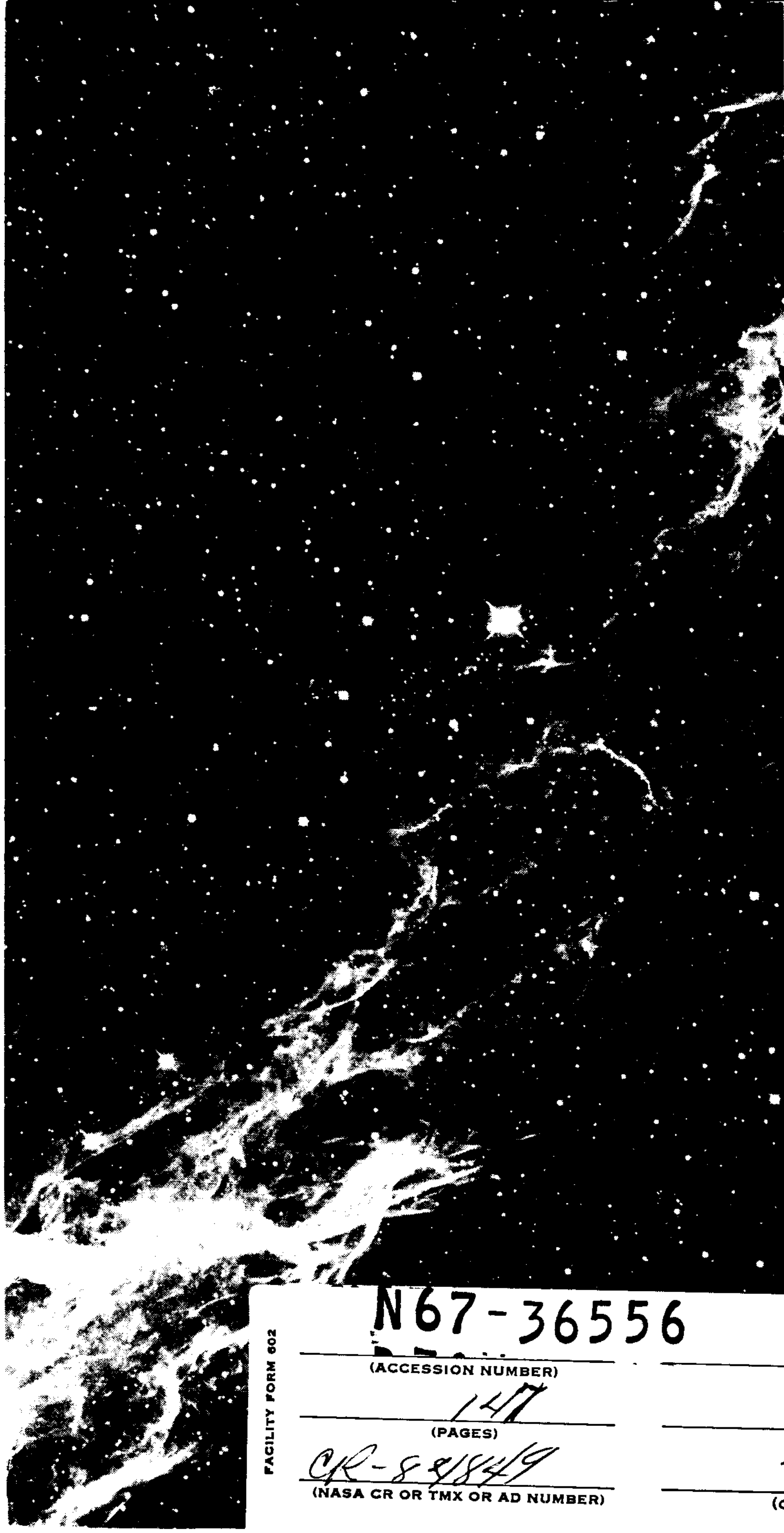

Report No. M-13

PRELIMINARY PAYLOAD ANALYSIS OF AUTOMATED MARS SAMPLE RETURN MISSIONS

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Compiled and Written
by
John C. Niehoff
Astro Sciences Center
of
IIT Research Institute
Chicago, Illinois

for

Lunar and Planetary Programs
Office of Space Science and Applications
NASA Headquarters
Washington, D. C.

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APPROVED:

C. A. Stone, Director
Astro Sciences Center

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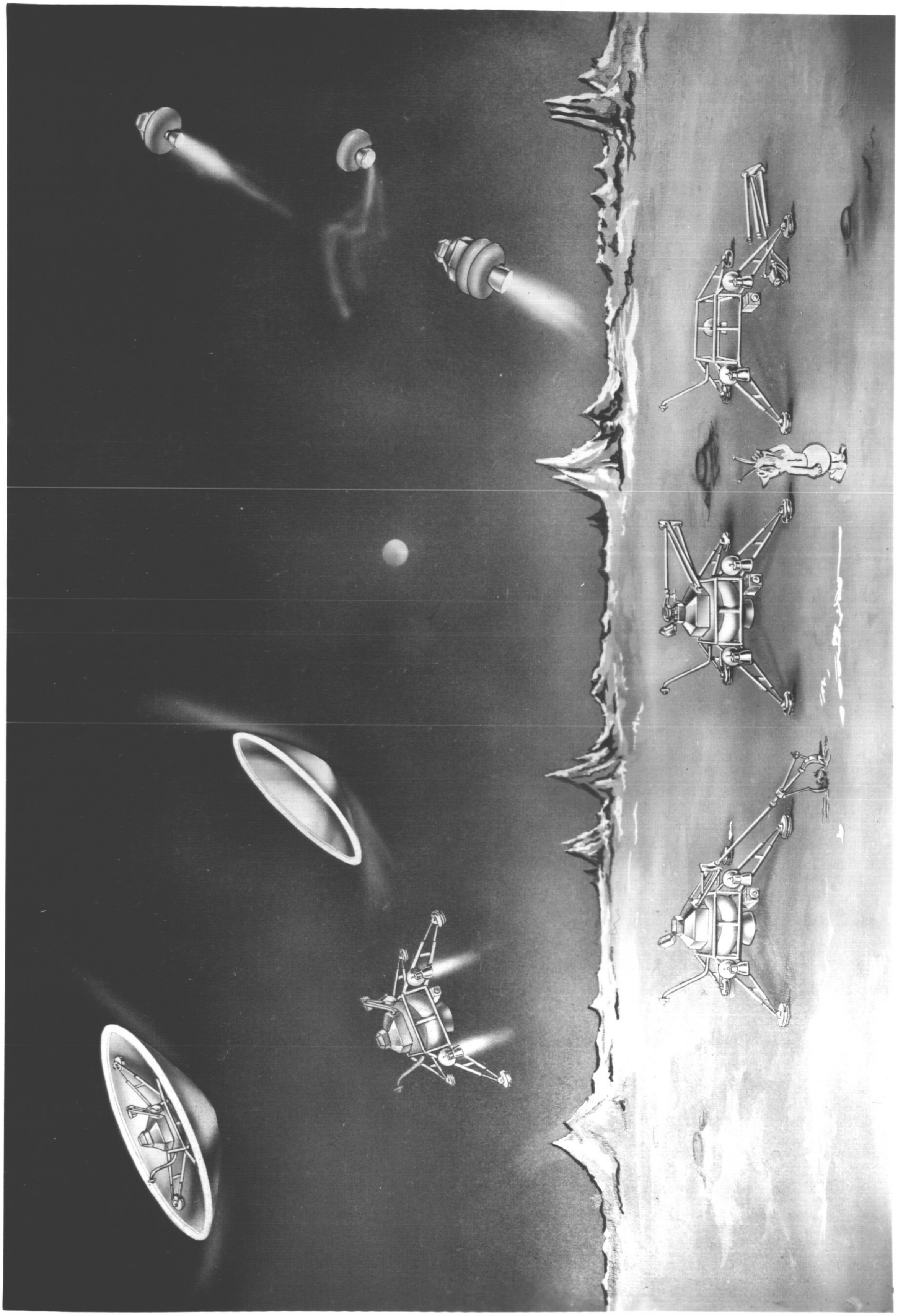
An illustration of the sequence of events involving automated surface sample collection at Mars if rendezvous in Mars orbit is employed.

From left to right the operations are initial atmospheric braking of the descent spacecraft, terminal powered descent to a soft landing, sample surveillance and collection, sample storage, launch, first stage burnout, and second stage ascent to rendezvous. For sample return without rendezvous the landed spacecraft may have to be enlarged to support a three-staged ascent vehicle.



FOREWORD

This report describes the results of a payload analysis study of Automated Mars Sample Return (AMSR) missions for the Lunar and Planetary Programs Office of NASA Headquarters. A concurrent study of the mission systems and design requirements was performed at JPL. The analysis is preliminary in nature due to a NASA quick-response requirement for presentation of results two months after initiation of the study. These results could not have been obtained within this two month period without a large measure of technical cooperation and discussion between JPL, NASA Headquarters, and IITRI.

In particular, Ken Fishback and Peter Feitis of JPL have provided timely information on the many systems and spacecraft configuration aspects of the mission study and have provided an invaluable cross-check on the trajectory calculations.

Frank Spurlock of the Space Flight Analysis group at NASA-Lewis Research Center has provided detailed calculations for the vehicle parameters and trajectories of the ascent phase at Mars. These results could not have been obtained without his enthusiastic support.

The following IITRI personnel have contributed to the study results:

J. Dockery	F. Narin
A. Friedlander	J. Niehoff
M. Hopper	D. Roberts
R. Lang	J. Waters

The frontispiece and graphical work are by Shawn Clark.

SUMMARY

Early Voyager missions to Mars may be followed by large automated biological laboratory (ABL) spacecraft designed to conduct *in situ* biological, geological and meteorological analyses on the Martian surface. Recently the possibility of returning a sample of the Martian surface back to Earth for analysis has gained interest as a complement to or substitute for the ABL mission. This report is concerned with the unmanned or automated collection of a sample of the Martian surface and its return to Earth. The mission is referred to as an Automated Mars Sample Return (AMSR). The study objective was to identify by preliminary analysis AMSR mission modes which could be launched in the mid-1970's by a single Saturn V vehicle if available chemical propulsion systems were used throughout the mission. The scientific justification for returning Mars samples to Earth was not considered in this study.

The study results are presented in three parts. Firstly, the mission is subdivided into key phases and several options are discussed for each phase. On the basis of study constraints and the restrained scope of interest (mid-1970 missions) enough relevant phase options are selected to formulate 12 candidate

mission modes. Secondly, the assumptions, supporting performance analyses, and the payload determination methodology are briefly discussed. Finally, four of the 12 candidate mission modes are shown to satisfy the study constraints and be within the capability of a single Saturn V launch.

These four mission modes are summarized in the Summary Table. Modes 1-3 all use minimum energy interplanetary transfers and a long (306 day) stay time at Mars. Total trip time is 975 days. Mode 1 is distinguished by a Mars capture orbit before descent and direct return of the collected sample from the Mars surface. Mode 2 employs a direct entry descent (no capture orbit) and direct return of the collected sample. Mode 3 uses a Mars capture orbit before descent and rendezvous in orbit before return of the collected sample to Earth. Mode 4 uses the same near-Mars options as Mode 3, but with higher energy interplanetary transfers and a Venus swingby (returning) to shorten the stay time to 12 days and the total trip time to 549 days. The required total spacecraft weight for each mode leaves a payload contingency of between 50 and 250 percent for a single Saturn V launch. However, a number of the assumptions which were made in order to determine these total weights require further verification and are specifically noted in the report conclusions.

AMSR mission feasibility appears to hinge on the demonstration of either of two new capabilities. These are

Summary Table

AMSR MISSION MODE SELECTION SUMMARY

	Mode 1	Mode 2	Mode 3	Mode 4
Trajectory Description				
Total flight time	975 days	975 days	975 days	549 days
Outbound flight time	364 days	364 days	364 days	200 days
Arrival and descent	via orbit	direct	via orbit	via orbit
Mars stay time	306 days	306 days	306 days	12 days
Launch & departure	via parking orbit	via parking orbit	via rendezvous in Mars orbit	via rendezvous in Mars orbit
Return	direct	direct	direct	Venus swingby
Return flight time	305 days	305 days	305 days	337 days
Earth reentry speed	37,650 ft/sec	37,650 ft/sec	37,650 ft/sec	39,500 ft/sec
Weight Summary				
Saturn V capability	75,000 lb	75,000 lb	75,000 lb	69,000 lb
Earth/Mars spacecraft	50,600	28,850	21,200	36,150
Mars entry capsule	22,300	26,750	7,100	7,100
Mars landed weight	8,920	8,920	3,550	3,550
Mars/Earth spacecraft	580	580	630	660
Earth reentry capsule	50	50	55	55
Key Features	Least complicated mission	Direct entry reduces initial S/C weight, but increases G&C requirements	Rendezvous before return reduces initial S/C weight, but increases G&C requirements	Rendezvous before & Venus swingby during return reduces initial S/C weight & stay time in exchange for most stringent G&C

Note: All Earth launches in September 1975. Weights in Earth lb.

- the ability to soft-land a 8-10,000 lb launch vehicle and ancillary equipment on Mars (Mode 1 and 2), or

- the ability to rendezvous and dock two unmanned spacecraft in Mars orbit (Modes 3 and 4).

A number of specific subjects are recommended for continued analysis in order to further determine the value and feasibility of AMSR missions. These subjects include:

- Determination of the scientific objectives applicable to AMSR missions,

- Evaluation of the sample size, and the collection and storage requirements,

- Analysis of Mars landing site availability and fluctuation with launch opportunity and Mars intercept options,

- Investigation of changing weight requirements and new mission modes (e.g., outbound Venus swingbys) with later launch opportunities,

- Analysis of rendezvous and docking schemes and associated systems design requirements,

- Definition of optimum descent profiles for heavy (8-10,000 lb) Mars landers,

- Comparison of assumed structure factors with available propulsion system hardware and designs,

- Analysis of midcourse requirements with different transfer trajectories (including Venus swingby) and Mars intercept options,

- Determination of payload penalties of launch windows and plane change requirements.

The study results support a continued interest in AMSR missions which should be encouraged at least until these recommendations are satisfied.

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1. INTRODUCTION

The purpose of the Voyager project is to return to Earth extensive photographic, biological, planetological, and meteorological data about Mars. Early missions will place spacecraft in loose orbits around Mars. Using multiple spacecraft modules these orbiters will send small capsules to the surface of Mars to conduct preliminary investigations. Later missions may send larger automated biological laboratories (ABL) to the Martian surface to conduct *in situ* analyses. Recently the possibility of returning a sample of the Martian surface back to Earth for analysis has gained interest as an addition to or substitute for the ABL mission.

Of particular interest in this study was the unmanned or automated collection of a sample of the Martian surface and its return to Earth. This mission is referred to as the Automated Mars Sample Return (AMSR) mission. The study objective was to identify several AMSR mission modes which would be feasible from a weight and energy standpoint using chemical propulsion systems expected to be available by the mid-1970's. It should be pointed out that at the beginning of this study it was not at all clear that any such mission modes existed. Also, no attempt was made to scientifically justify the return of a single, automatically procured Mars sample to Earth.

The interest in application of the AMSR mission to exploration of Mars at a relatively early opportunity (1975-1980) coupled with a quick-response requirement resulted in the following study constraints:

- Launch with a single Saturn V or smaller vehicle,
- Use the 1975 and 1977 Mars opportunities,
- Employ Mariner, Surveyor, and Voyager technology to the maximum extent practicable,
- Restrict the Mars sample size to 1-2 lb,
- Base the Mars atmospheric descent analysis on the worst case (VM-8) Mars model atmosphere,
- Use sterilizable propellants below a 1,000 kilometer altitude at Mars,
- Limit Earth reentry speed to 45,000 ft/sec or less.

A concurrent study of systems requirements of the AMSR mission, conducted at the Jet Propulsion Laboratory, determined that a 50 lb Earth reentry capsule would have to be carried throughout the AMSR mission to return to Earth the 1-2 lb sample of Mars. This and several other hardware weights determined in the JPL study are the basis for subsequent weight statements presented in this report.

2. MISSION OPTIONS

To separate relevant mission options from the myriad possibilities which exist the mission profile was divided into distinct phases. The phases chosen were:

Earth launch,
Earth/Mars transfer,
Mars intercept,
Atmospheric entry, descent, and landing,
Launch and ascent,
Mars/Earth transfer,
Earth intercept, and
Recovery.

Options were then selected for each phase with particular regard to study constraints and the scope of interest. Finally, the complete array of options was combined into a flow chart which would permit plausible mission mode selections. This chart is presented in Figure 1. The discussion which follows describes each set of phase options in moderate detail. The elimination of several options is explained on the basis of preliminary analysis and the study constraints.

2.1 Earth Launch

The only option considered for the Earth launch phase was a single Saturn V launch using a 100 N.M. parking orbit.

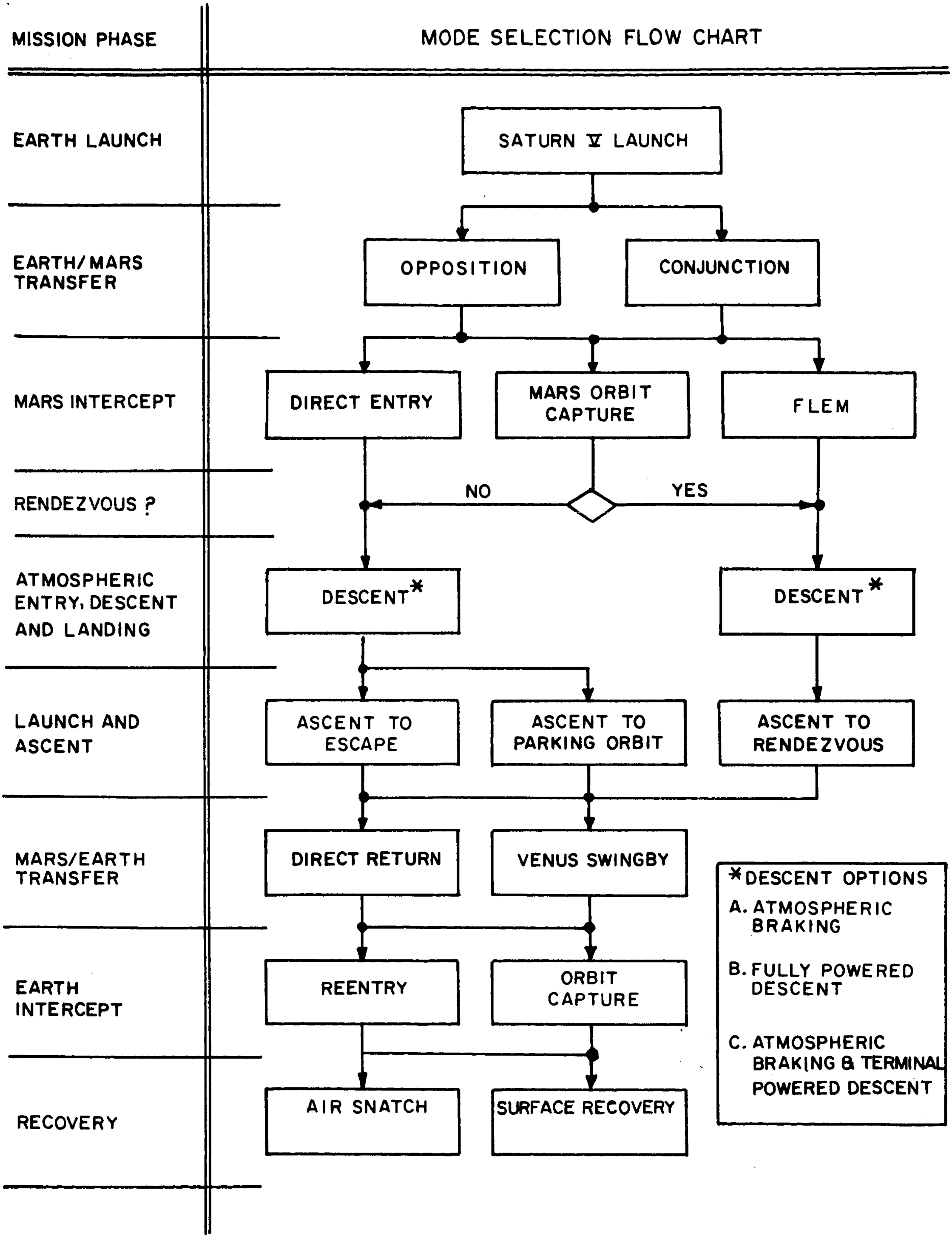

FIGURE I. OPTION ARRAY - INITIAL PHASE OPTIONS

2.2 Earth/Mars Transfer

The Earth/Mars transfer phase has, theoretically, an infinite number of trajectory possibilities. For this study the area of interest was confined to transfers which have arrival dates during the 12 month period between Earth/Mars opposition and conjunction. The two options shown in Figure 1, opposition and conjunction, portray explicit differences as the arrival dates approach these limits. The differences which characterize each option can be listed qualitatively as follows:

Opposition

High-energy transfer
high Earth escape velocity and
high Mars approach velocity

Trip time less than 9 months

Minimized communication distance

Conjunction

Low-energy transfer
low Earth escape velocity and
low Mars approach velocity

Trip time greater than 1 year

Solar blackout of communications
at arrival.

For Earth/Mars, absolute minimum energy outbound transfers arrive at Mars 3-5 months before conjunction. These are usually more attractive than true conjunction transfers since less energy is required and communication blackout does not occur until approximately two months after arrival. Hence, where reference is made in this and succeeding sections to conjunction transfers, absolute minimum energy trajectories are in fact implied. The term conjunction, although now used in a rather loose sense, is retained to differentiate this option from the higher energy opposition transfer.

The dominant parameter to be considered in the selection of the Earth/Mars transfer for a round-trip Mars mission is stay time at Mars.* The required stay time, in turn, emerges from the characteristics of the Mars/Earth return transfer which is selected. Mars/Earth transfer options are considered in Section 2.7 and only the resulting effect of short and long stay time will be stated at this time.

For short stay time (several weeks) Mars arrival dates must be close to opposition so that neither the return transfer energy nor the Earth reentry speed become excessive. For long stay time (10-12 months) Mars arrival dates closer to conjunction can be used while still retaining low energy return transfers and low Earth reentry speed.

2.3 Mars Intercept

Three options were given preliminary consideration for the Mars intercept phase. The first two options are direct entry into the Martian atmosphere and orbit capture. FLEM* (Titus 1966), is the third option. With this option a flyby spacecraft ejects a landing capsule about a week before Mars

*A number of detailed reports have discussed at length the trajectory/stay time trade-offs of round-trip missions, e.g., Sohn (1966), and Ehricke (1963).

**Flyby-Landing Excursion Mode.

intercept. The capsule enters the Martian atmosphere somewhat ahead of the spacecraft which is still traveling on a hyperbolic flyby trajectory. Several hours after landing the capsule is launched to perform a hyperbolic rendezvous with the spacecraft as it flies by. The spacecraft may be deflected during flyby so that after recovery of the capsule it continues with little or no additional propulsive velocity change to an Earth intercept. Some of the more important aspects of each option are:

<u>Orbit Capture</u>

Least critical approach guidance requirement

Landing site selection possible from orbit

In orbit storage during communication blackout (useful with Earth/Mars conjunction transfers)

Lower atmospheric entry velocity

Highest characteristic velocity requirement

Rendezvous available but not necessary

Variable stay time

<u>Direct Entry</u>

Critical approach and entry guidance and control requirements

No orbiter site selection

Higher atmospheric entry velocity

Low characteristic velocity requirement

Rendezvous not possible

Variable stay time

FLEM

Critical approach, entry and rendezvous guidance/control requirements

Critical launch timing requirements

No orbiter site selection

High atmospheric entry velocity

Low characteristic velocity requirement

Hyperbolic rendezvous necessary

Short stay time.

The FLEM option had to be eliminated, primarily because sufficient parametric data were not immediately available for analysis. However, orbit capture and direct entry should probably be considered separately since neither of these options have the necessary short stay time and short return launch window required for the FLEM. The lower total mission weight requirement* expected if FLEM were used, would thus appear attractive only if total weight requirements of the other intercept options exceed the capability of the single Saturn V launch constraint.

2.4 Rendezvous

At this point in the selection of mission options the decision must be made whether or not to use rendezvous at Mars. If a direct entry has been selected then, of course, rendezvous

*Lower total mission weight is the result of not having to capture the interplanetary and Earth reentry hardware at Mars.

is not possible.* But if orbit capture is selected, a significant reduction in total payload weight is possible when rendezvous is used. The propulsion and systems required to rendezvous and bring the sample back to Earth remain in orbit. Only the mechanisms for collecting, storing and launching a sample need be sent to the surface. After sample collection this much lighter payload is launched to a parking orbit prior to rendezvous with the orbiting return module. The reduced descent/ascent payload is gained, of course, at the expense of increased systems complexity. The Mars orbit capture option was considered with and without rendezvous in the mission mode selections.

2.5 Mars Atmospheric Entry and Landing

Three options were postulated for braking the atmospheric entry capsule to a soft Mars landing. As shown in Figure 1 they are (1) complete atmospheric braking including a parachute for terminal descent, (2) powered (rocket) descent disregarding the atmosphere as a significant energy dissipator, and (3) initial atmospheric braking combined with a terminal powered descent.

A recent study of Mars atmospheric probes and landers (AVCO, May 1966) concluded that a maximum weight of 600 lb could be successfully landed on Mars using atmospheric braking (without

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*Direct entry can be combined with rendezvous if a split option is used for the Mars intercept phase. That is, the landing capsule may enter the atmosphere directly if it is separated before the Earth return vehicle enters Mars orbit. This combination of options was not analyzed due to the limited time available for the study.

lift) and a terminal parachute descent. An impact velocity of 130 ft/sec required that 30 percent of the landed weight be used for impact attenuators in order to meet a maximum deceleration design specification of 500 g. These results are based on a minimum Mars surface pressure of 10 millibars.

For a number of reasons it was clear that complete atmospheric braking (terminal parachute descent) had to be ruled out as a plausible descent option. Preliminary evaluation of landed weight requirements to launch a 1-2 lb surface sample (and related hardware) back into Mars orbit indicated a range of 2500-3500 lb, four to six times the maximum weight of 600 lb cited above. The high impact velocity and attendant deceleration loads were also judged incompatible with the moderately complex launch vehicle system which must be landed. Although not specifically investigated, it was felt that a Surveyor-type soft landing (impact velocity $\leq$ 25 ft/sec) would be a much more reasonable touchdown constraint for eventual capsule design. Finally, as a result of Mariner IV data, the Mars surface pressure has been reduced to a range of 5-10 millibars from the minimum of 10 millibars used in the AVCO results. This lower minimum surface pressure would result in even higher impact velocities.

The second descent option is a powered (rocket) descent disregarding the atmosphere of Mars as a means of energy dissipation. The descent performance of rocket braking can be expressed as the ratio of entry weight to landed weight.* This ratio was investigated for direct (hyperbolic) approaches to the planet assuming rectilinear motion and a constant Mars gravity field. The ratio varies with the maximum deceleration load (G-LOAD) experienced during descent. A curve of weight ratio versus G-LOAD is shown in Figure 2. The ignition velocity (V_o) of 5.85 km/sec corresponds to a hyperbolic approach velocity of about 3 km/sec. A terminal boundary condition of zero velocity at zero altitude was assumed. The performance ratio is based on a two-stage descent using a specific impulse (ISP) of 310 sec and a propellant sensitive propulsion hardware factor (f) of 15 percent for both stages.

Assuming that a maximum deceleration load of 25 Earth g's can be tolerated** then from Figure 2 the resulting ratio of entry weight to landed weight is 12.9:1. This ratio is prohibitively expensive. If an assumed 10,000 lb soft-lander can return a 1-2 lb sample of Mars directly (without rendezvous) back to Earth then the descent spacecraft would have to have an initial weight of 129,000 lb. This is almost twice the Mars mission payload capability of a single Saturn V launch vehicle.

The compromise solution to the Mars descent phase considered in this study was option three - initial atmospheric

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*The term landed weight, as used throughout this report, includes neither descent propulsion hardware weight nor aeroshell weight (used in the third descent option).

**The maximum G-LOAD on Surveyor I during its retro and descent maneuvers was only 9.7 g (JPL 1966a).

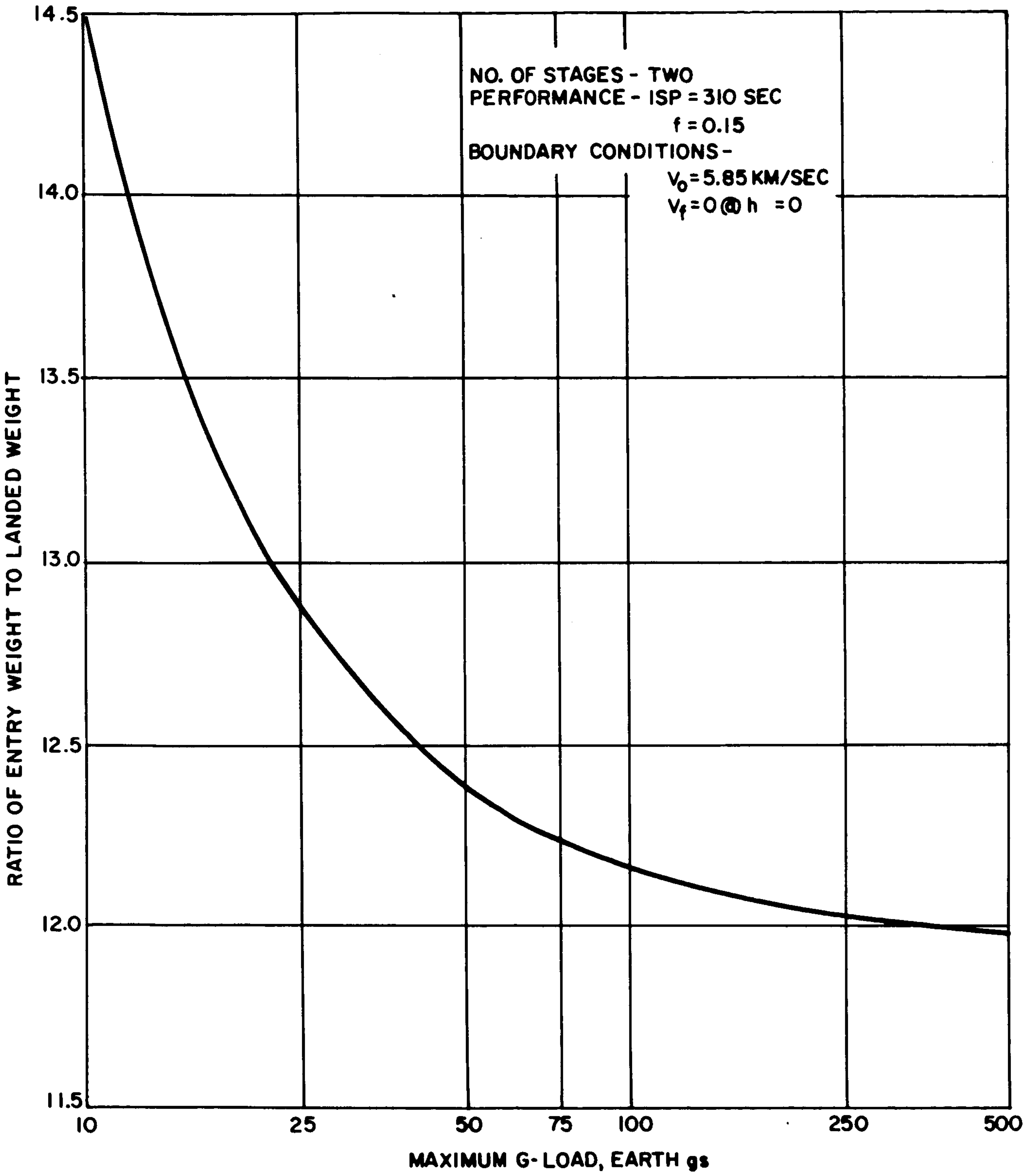


FIGURE 2. WEIGHT RATIO FOR DIRECT-ENTRY FULL-POWERED DESCENT

braking followed by terminal powered descent. Atmospheric braking in the early portion of the descent improves the descent performance while terminal powered descent insures a soft landing. Preliminary evaluation of the performance was expressed in terms of entry weight to landed weight ratios. A detailed account of the discussion which follows is given in the Appendix, Section A.4.

Overall descent ratios for an out-of-orbit approach are plotted in Figure 3 as a function of ballistic coefficient (B) for a constant ignition altitude of 5 km. A constant scale height (10 km) 5 mb surface pressure model was assumed to simulate the Mars atmosphere. Entry altitude was set at 50 km (164,000 ft). The entry velocity is 4 km/sec and the entry flight path angle is -15°. The aeroshell weight was assumed to be a constant 30 percent of the entry weight. An ISP of 310 sec and a propellant hardware factor of 0.15 were used for the terminal propulsion stage. The variation in maximum landed weight with ballistic coefficients is also shown in Figure 3. At any given value of B there is an upper limit to the capsule entry mass, M_e, if the aeroshell diameter is constrained by the launch vehicle shroud dimensions or design integrity constraints. Since the standard Saturn V shroud diameter is 20 feet, the aeroshell diameter was assumed to be limited to 19 feet. Using a drag coefficient (C_d) of 1.4 and B = 1.0 would limit the maximum entry mass to 397 slugs or about 12,800 Earth lb. With the descent ratio of 2.57 at B = 1.0 from Figure 3 the maximum

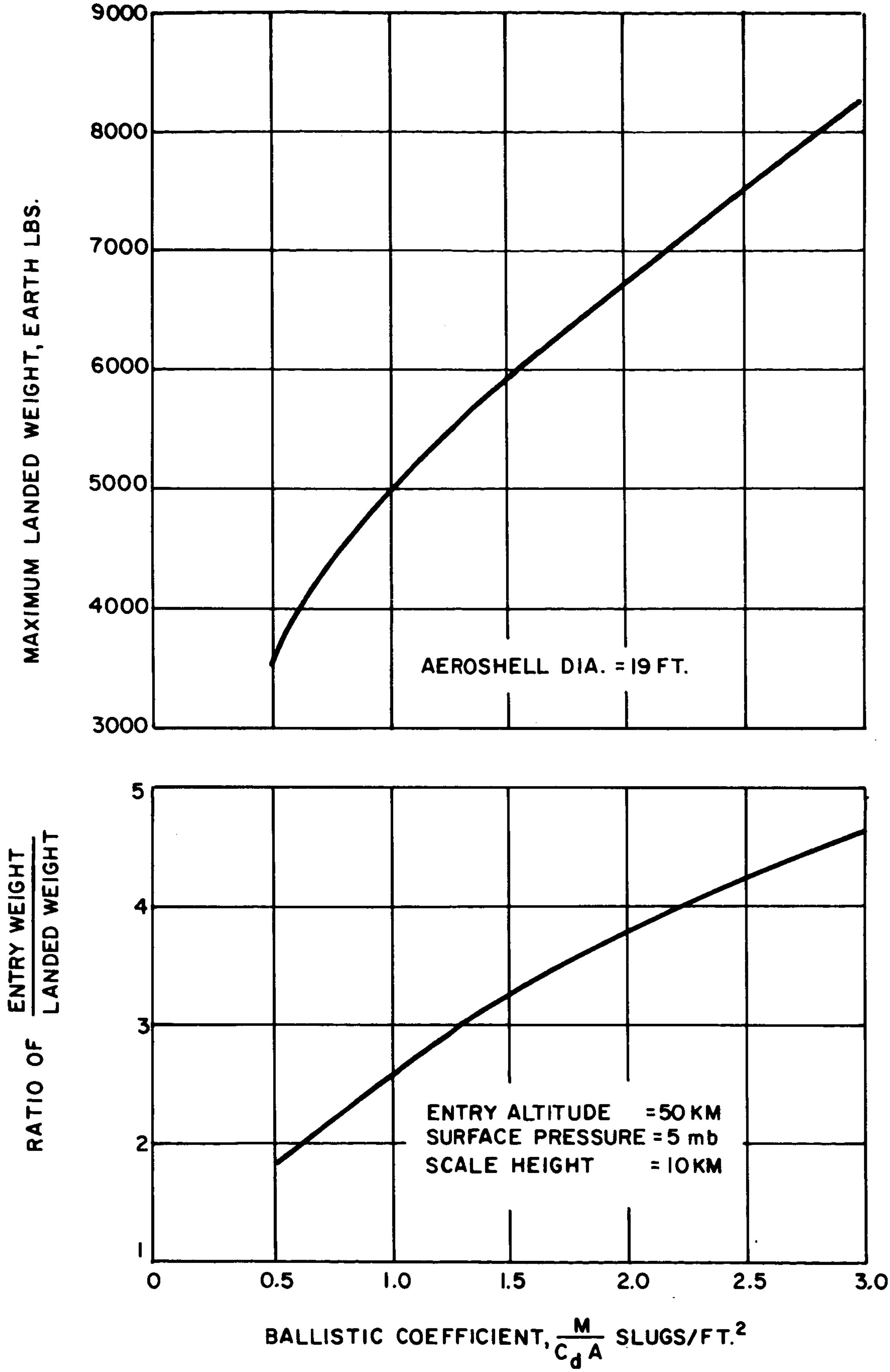


FIGURE 3. PRELIMINARY OUT-OF-ORBIT DESCENT PERFORMANCE

landed weight would be 12,800/2.57 = 4980 lb. A similar presentation of descent ratios and maximum landed weights as a function of ballistic coefficient for direct (hyperbolic) entry is presented in Figure 4. The specific differences in the data of Figure 4 are a higher entry velocity of 5.85 km/sec and a steeper entry angle of -20° (to avoid skip-out at the higher entry velocity).

A number of preliminary comparisons for combined atmospheric braking and terminal powered descent can be drawn from Figures 3 and 4. Firstly, the descent ratios, which are a measure of performance, are significantly better (lower) than complete powered descent for ballistic coefficients less than about 1.5.

Secondly, there are two opposing factors which affect the selection of ballistic coefficient. The descent ratio improves as the ballistic coefficient <u>decreases</u>. The maximum landed weight increases as the ballistic coefficient <u>increases</u>. Hence the selection of ballistic coefficient should be made such that the required landed weight is equal to the maximum value. This would guarantee the best descent performance to land the required weight. However, as heavier landed weights are required the increase in the entry capsule weight (and ultimately total spacecraft weight) is magnified by an associated loss in performance. Therefore, it is desirable to avoid ever increasing the ballistic coefficient to accommodate heavier

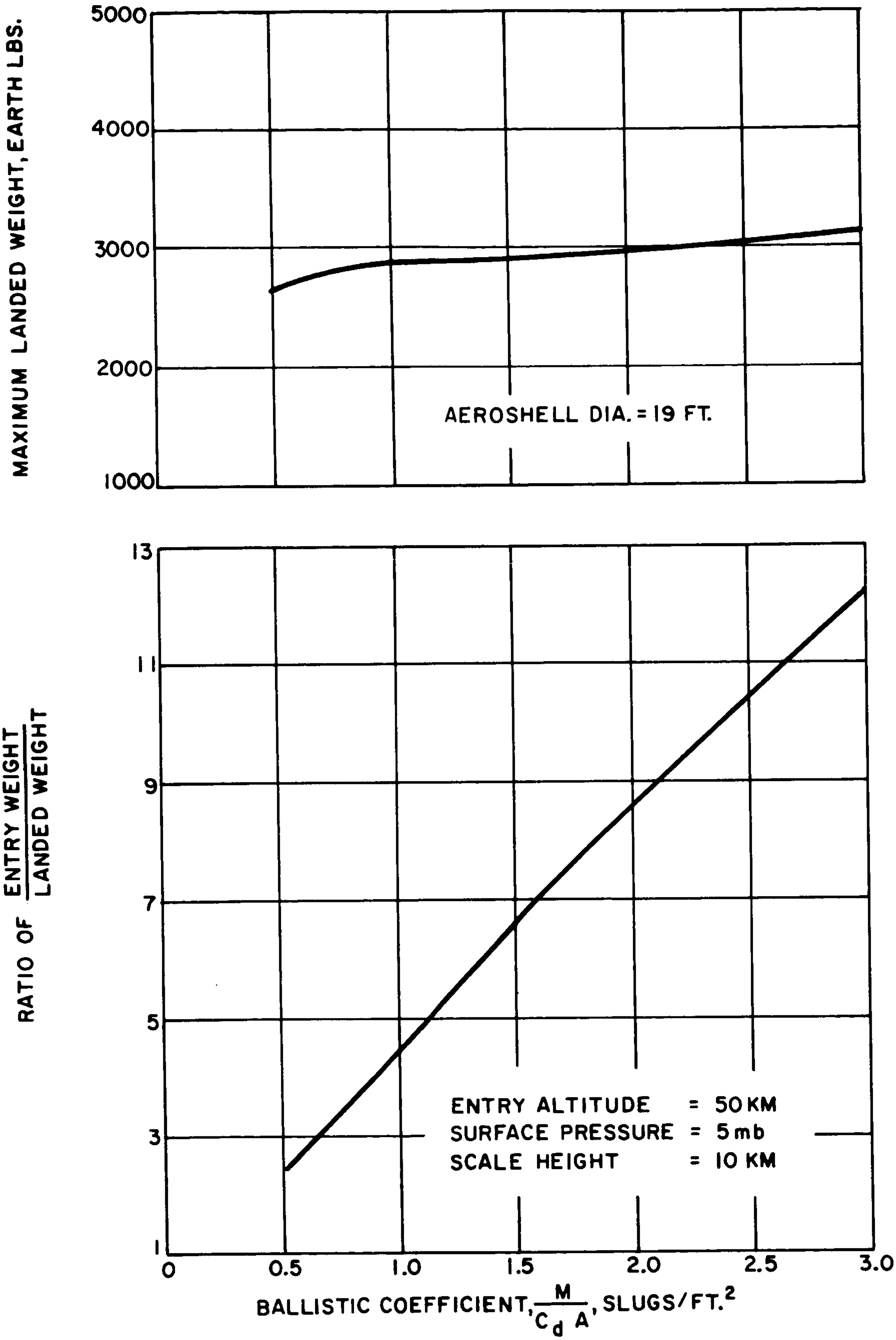


FIGURE 4. PRELIMINARY DIRECT ENTRY DESCENT PERFORMANCE

landed weight, meaning that at some point the constraint of maximum aeroshell diameter must be considered and perhaps changed.

Thirdly, for direct entry (Figure 4) the curve of maximum landed weight for a 19 foot aeroshell shows very little variation with ballistic coefficient. While a lesser variation is expected due to the more rapid increase in the descent ratio curve, that effect has been exaggerated by the use of a -20° entry angle at the low entry altitude of 50 km (164,000 ft). In fact later more detailed trajectory analysis revealed that an angle of -12° at 164,000 ft is equivalent to an angle of -20° at 800,000 ft (vacuum entry altitude).

A number of considerations have been omitted from this preliminary analysis which are essential to selection of optimized descent profiles. Among them are optimized ignition altitude, variable aeroshell diameters and weight fractions, propulsion system reaction time for terminal descent and maximum deceleration loads to name a few. A more accurate analysis of entry trajectories was performed later in the study which permitted the selection of representative descent ratios for payload tabulation. However, trade-off analyses leading to optimized descent profiles were beyond the scope of this study.

2.6 Mars Launch

Three options were considered for launching a return sample from the Mars surface. The first two, direct ascent to escape and ascent to parking orbit, apply to non-rendezvous configurations. The ascent to escape option is a launch from the surface directly onto a hyperbolic escape trajectory. The parking orbit option was preferred to the direct ascent option for two reasons. Firstly, a parking orbit simplifies the launch requirements by fixing the ascent geometry to orbit and secondly, if long stay times following landing are required, the time could be spent in orbit avoiding long term storage of the launch vehicle system and propellant on the Martian surface. The surface is probably a more hostile environment than that encountered in Martian orbit. No engine restarts would be necessary if a three stage system (to escape) is used, since parking orbit insertion occurs at the second stage burnout. Hence the direct ascent to escape option was eliminated as a launch option.

The third option, ascent to rendezvous, applies only to mission modes which include rendezvous. It is similar to the ascent to parking orbit profile but with a slightly higher energy requirement. This is due to the fact that the rendezvous parking orbit had to be raised to avoid contamination of Mars by the unsterilized orbiting bus during rendezvous.

2.7 Mars/Earth Transfer

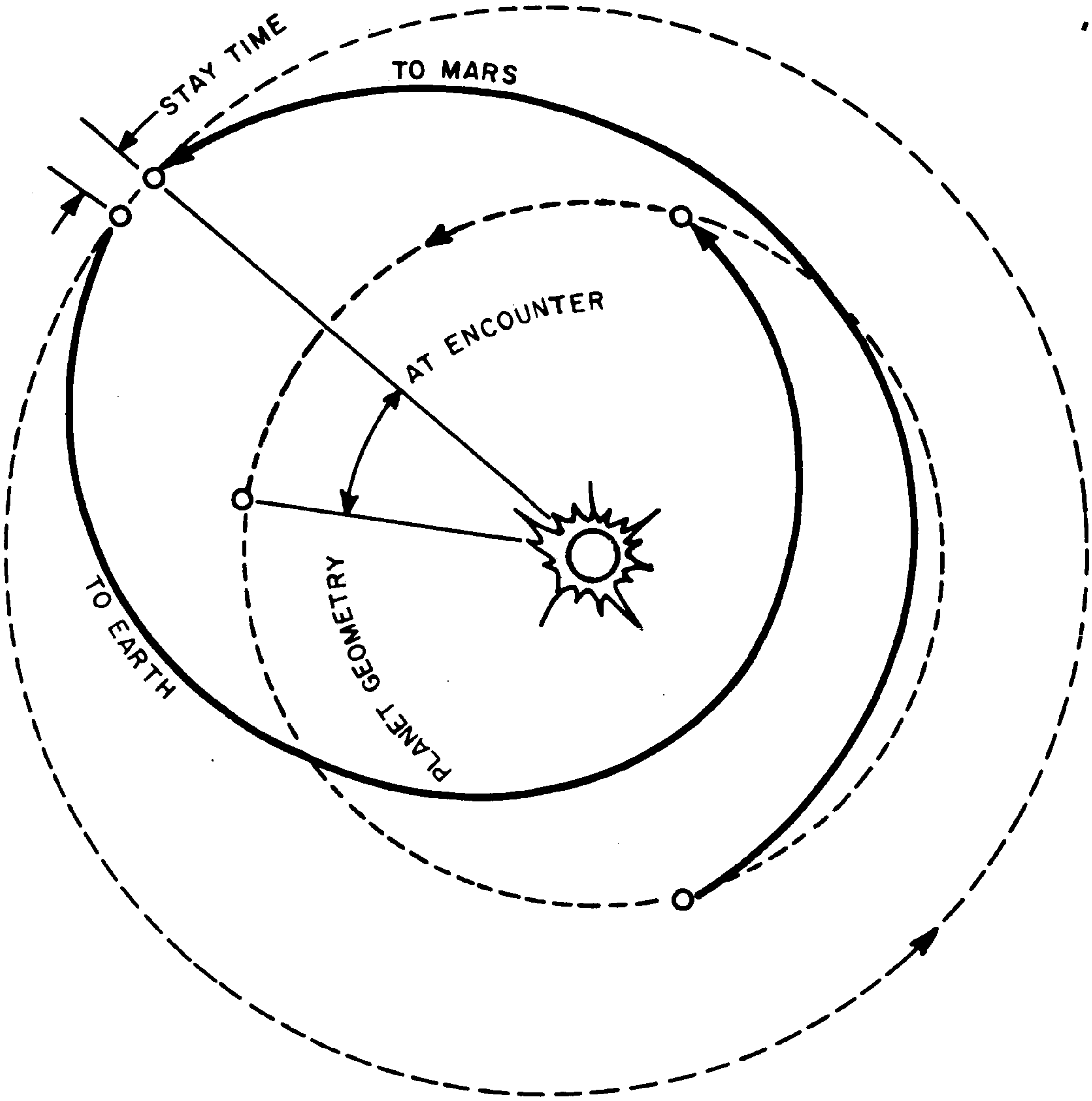
Two options, direct return trajectories and Venus swingby return trajectories, were shown in Figure 1 for the Mars/Earth transfer phase. Two constraints were imposed on this phase of the mission, (1) that all return transfers considered would be minimum energy trajectories, and (2) that acceptable return transfers have entry velocities less than

45,000 ft/sec. The first constraint guaranteed a minimum escape dV requirement from Mars orbit on any given day. The second constraint reflects a preliminary estimate of Earth reentry capability for a 1975 mission and permitted an estimate of the Earth reentry capsule weight.

Mars launch dates which were considered for the return transfer options, began with the earliest possible arrival date (Mars opposition, December 15, 1975) and varied over a period approximately equal to one complete revolution of Mars about the Sun - 22.5 months. Trajectories for the two return options considered were found to satisfy the imposed constraints on very different Mars launch dates. Hence, their combination with the outbound transfer options formed very different Mars stay time mission modes.

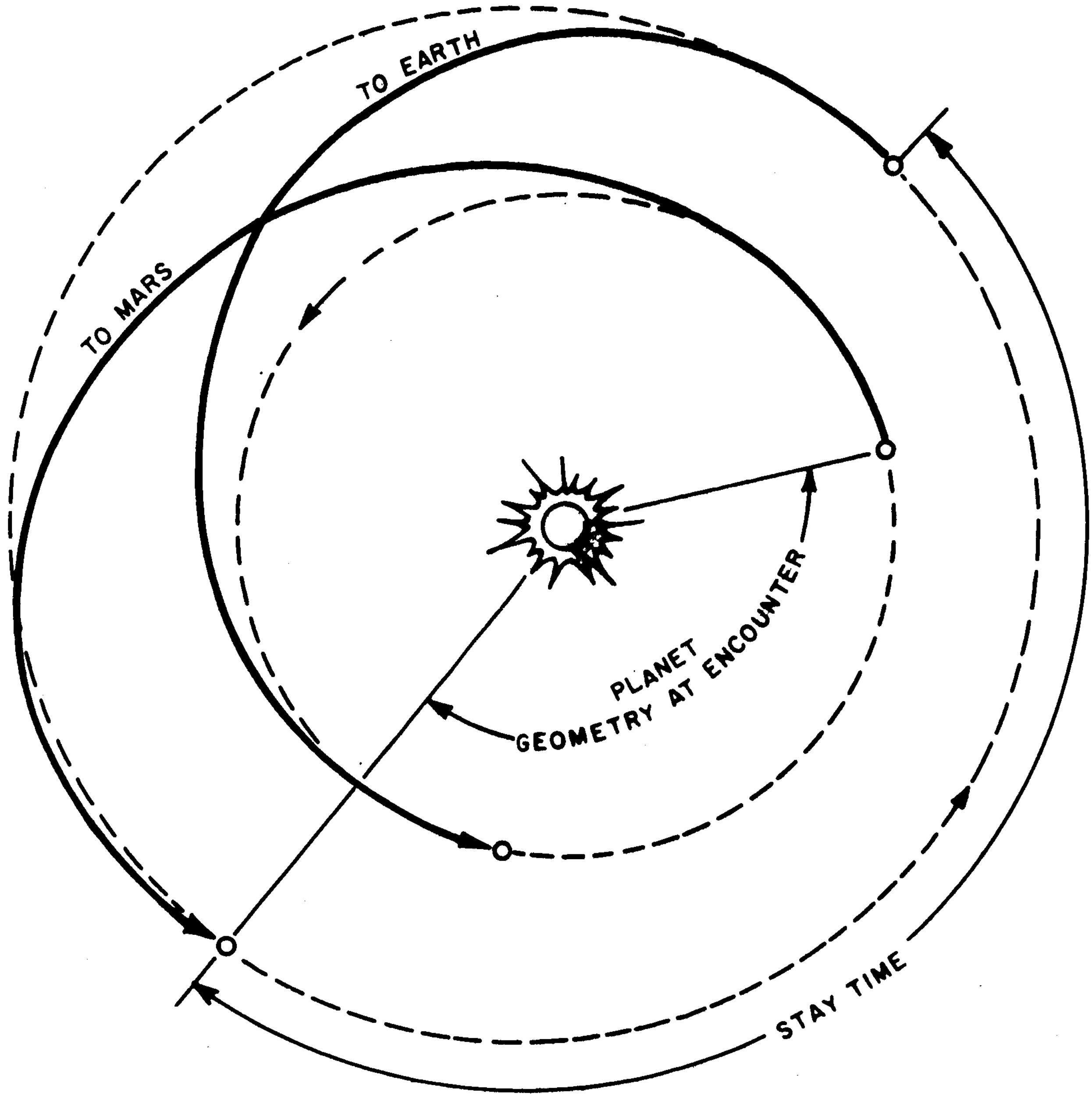
For the direct return transfer option, minimum energy trajectories with minimum Earth reentry speeds (less than 40,000 ft/sec) occur about 5 months <u>before</u> Mars opposition. By Mars opposition (the earliest Mars arrival date considered) minimum return trajectories have Earth reentry speeds already in excess of 45,000 ft/sec. An even further delay in the Mars launch date in order to improve the payload on the <u>outbound</u> transfer (see Section 2.2) and provide a reasonable Mars stay time after arrival of at least 10 days not only increases the Earth reentry speed but also the escape velocity requirement at Mars for a minimum return trajectory. These problems are basically the result of the unfavorable position of the Earth at the time the spacecraft encounters Mars. To illustrate this the heliocentric transfer geometry of an opposition type round-trip mission has been plotted in Figure 5. The Mars arrival date is about two months after opposition. The stay time at Mars is only 10 days. Due to the position of the Earth at Mars encounter, the spacecraft spends the entire return transfer catching up with the Earth, completing more than three-quarters of a revolution about the Sun in doing so. When the spacecraft finally reaches Earth, its path cuts across the Earth's orbit, which results in a high reentry speed of almost 65,000 ft/sec. The only advantage gained by this opposition type mission is a relatively short mission time of about 520 days.

To reduce the Earth reentry speed of direct return transfers below the 45,000 ft/sec constraint the spacecraft must remain at Mars until the Earth reaches a more favorable position in its orbit. From a Mars arrival date at opposition this stay time would be more than one Earth year. However, using lower energy conjunction-type outbound transfers with Mars arrival dates some seven months after opposition reduces the stay time and increases the payload at Mars at the same time. The heliocentric transfer geometry of a conjunction type round-trip mission is illustrated in Figure 6. The outbound transfer has been selected to maximize the payload at Mars encounter. The stay time is 306 days. The minimum energy direct return transfer shown yields an Earth reentry speed of less than 40,000 ft/sec. This is characteristically a near-minimum energy mission. Its



MARS ARRIVAL DATE - - - - - - - - - - - - FEBRUARY 13, 1976
MARS STAY TIME - 10 DAYS
EARTH REENTRY SPEED - - - - - - - - - - - - 64,800 FT/SEC
TOTAL MISSION TIME - - - - - - - - - - - - - - - - - - 520 DAYS

FIGURE 5. TRANSFER GEOMETRY OF OPPOSITION TYPE MISSION



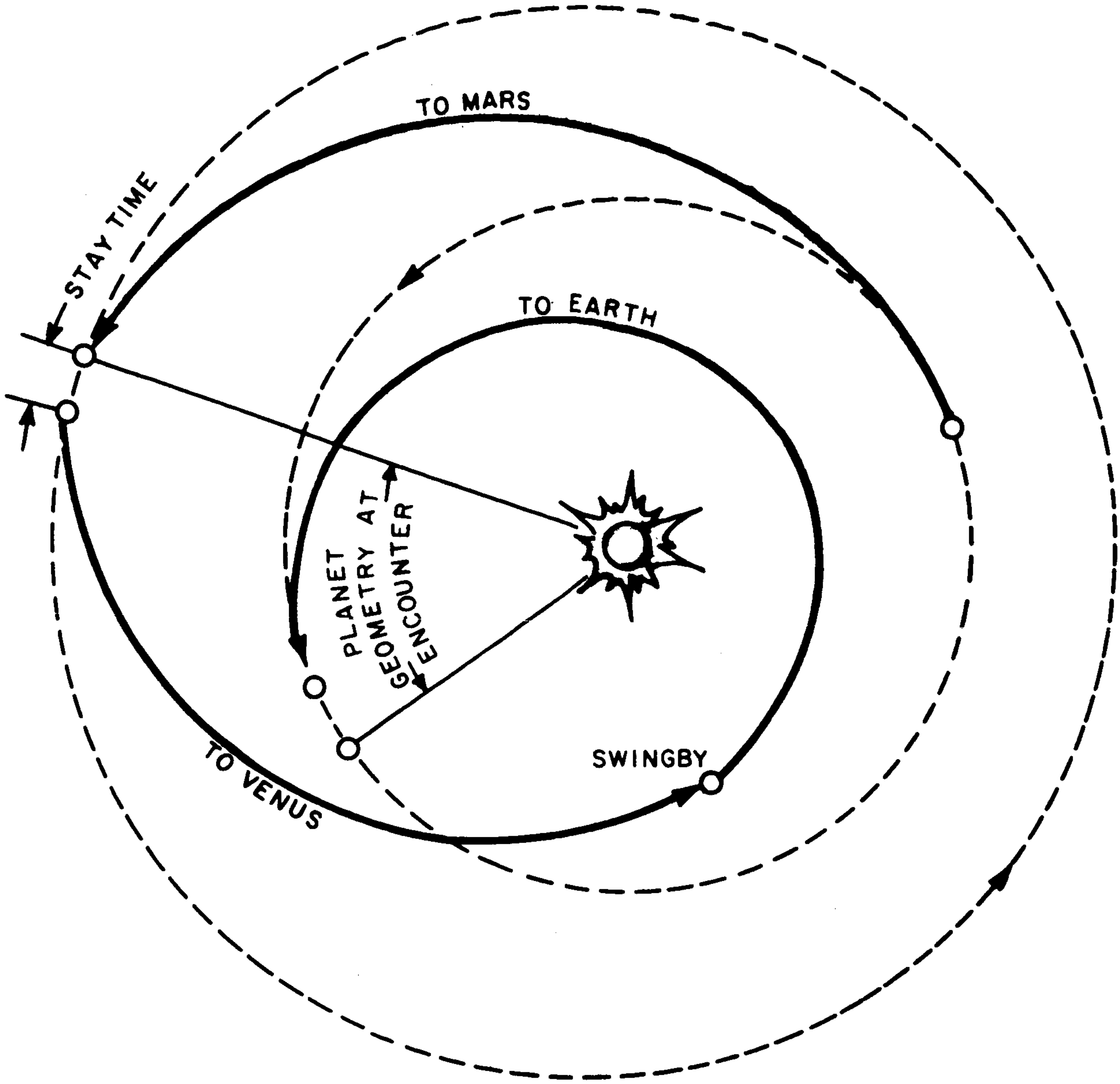
MARS ARRIVAL DATE	SEPTEMBER 7, 1976
MARS STAY TIME	306 DAYS
EARTH REENTRY SPEED	37,650 FT/SEC
TOTAL MISSION TIME	975 DAYS

FIGURE 6. TRANSFER GEOMETRY OF CONJUNCTION TYPE MISSION

primary disadvantage is a long mission time of nearly 1000 days (2.7 years).

The combination of direct outbound transfers with direct return transfers results in either of two problems for round-trip Mars missions - high Earth reentry speed or long total mission time. The addition of a Venus swingby maneuver to either the outbound or return transfer (depending upon the launch year) will almost always eliminate both of these problems (Deerwester and D'Haem 1967). For the 1975 launch opportunity, which was primarily used in this study, the Venus swingby is part of the return transfer. This was the second option of the return transfer phase shown in Figure 1. For other launch years the Venus swingby could appear instead as an outbound transfer phase option.

To illustrate the effectiveness of the Venus swingby maneuver in avoiding the problems of long trip time and high Earth reentry speed, the heliocentric transfer geometry of an opposition type round-trip Mars mission mode using the Venus swingby return transfer option is shown in Figure 7. Mars encounter occurs about 4-1/2 months after opposition. The Earth's position at encounter is even worse than it was for the first example shown in Figure 5. But now, in the process of catching up with the Earth on the return transfer, Venus alters the spacecraft's orbit so that it approaches the Earth nearly tangentially at intercept. The resulting reentry speed is less than 40,000 ft/sec. Since more than one complete revolution



MARS ARRIVAL DATE - - - - - - - - - - - - - - - APRIL 1, 1976
MARS STAY TIME - 12 DAYS
ALTITUDE OF CLOSEST VENUS APPROACH - - - - - - - - 1130 KM
EARTH REENTRY SPEED - - - - - - - - - - - - - - 39,600 FT/SEC
TOTAL MISSION TIME - 549 DAYS

FIGURE 7. TRANSFER GEOMETRY OF OPPOSITION TYPE MISSION WITH VENUS SWINGBY RETURN.

around the Sun is made in catching up with the Earth, the total mission time of 549 days is slightly longer than 520 days for the opposition type mission in Figure 5, but still far below the mission time of 975 days for the conjunction type mission in Figure 6.

An additional advantage of adding a returning Venus swingby to the opposition type round-trip mission is a lower total energy requirement resulting from the later arrival date at Mars. This, of course, means improved payload capability of a fixed launch vehicle system, in this case the Saturn V. The specific disadvantages which attend the Venus swingby and must also be considered are the added guidance requirements of an accurate Venus flyby and in some cases approaches to within 0.5 AU of the Sun.

2.8 Earth Intercept

The collected sample in a reentry capsule can be guided directly into the Earth's atmosphere and braked aerodynamically, or the spacecraft can perform a retro capture maneuver to place the sample in a loose Earth capture orbit where it could later be retrieved by another Earth launched spacecraft (possibly manned). The first option is certainly simpler and less expensive. However, the assumed reentry capsule (weight = 50 lb) cannot sustain heating for reentry speeds above 45,000 ft/sec. Hence retro propulsion would be necessary to either slow the capsule down before entry or place it in a loose capture orbit (the second option)*. This trade-off was not resolved in the study and because reentry speeds of less than 40,000 ft/sec were usually possible only the first option was considered for mission mode selections.

2.9 Earth Sample Recovery

Since the sample size returned from Mars was constrained to 1-2 lb the required reentry capsule should be small in size and weight. Recovery of the reentered payload from the Earth's surface (most likely an ocean) is less attractive than the option of an air recovery. An air snatch was assumed as the primary recovery technique with surface recovery included as a backup.

2.10 Option Summary

Mission configurations could be generated from Figure 1 by selecting one option per phase and proceeding vertically from launch to recovery along the phase connected lines. This would give 264 possible mission modes. Figure 8 is a revision of Figure 1 showing the options which have been eliminated. Only 12 valid candidate mission modes remain if the two recovery alternatives shown are considered as one option.

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*There is also the possibility of increasing the ablative material on the reentry capsule, but a motivation to minimize the capsule weight and lack of sufficient analysis in this area ruled out this consideration in the study.

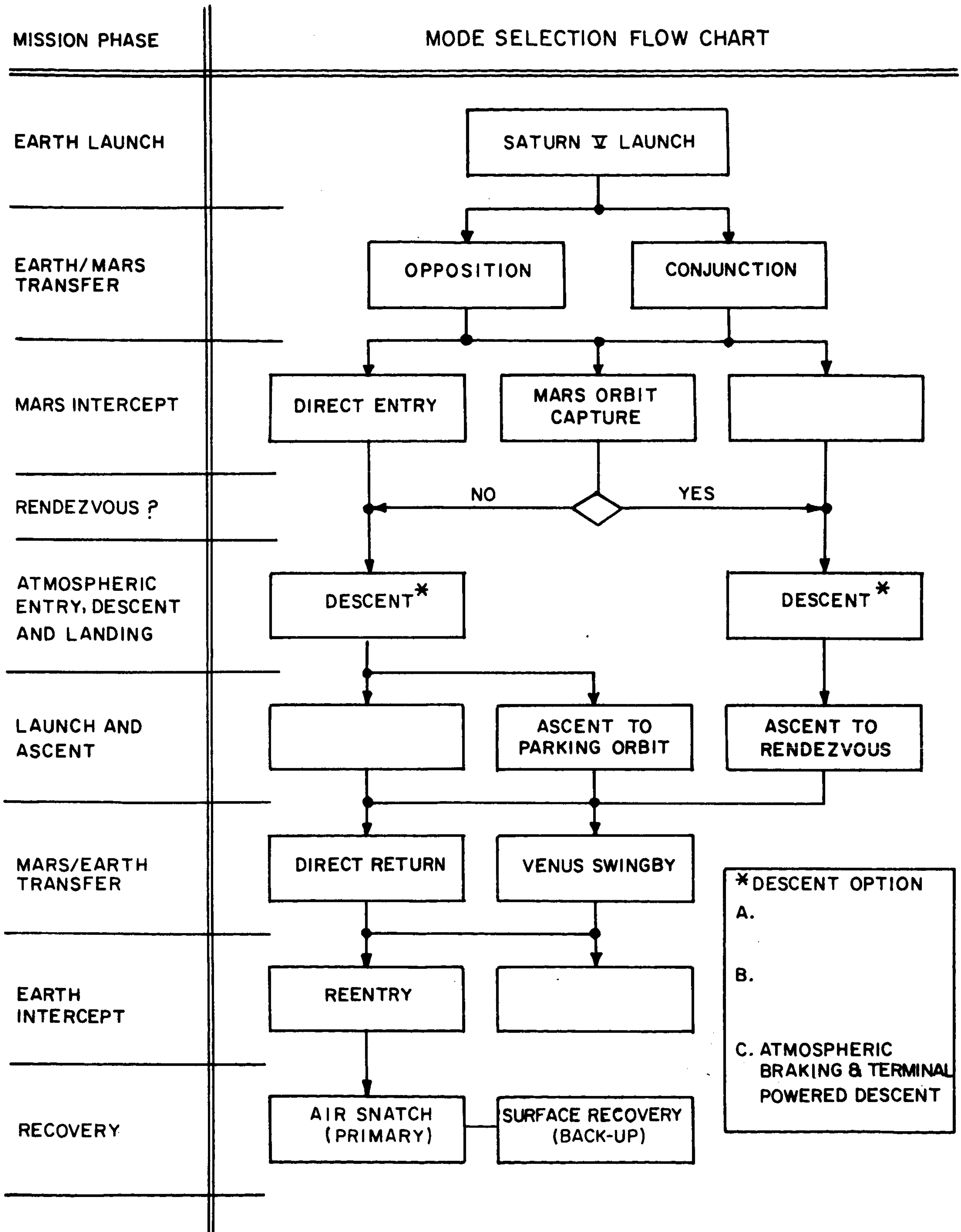

FIGURE 8. OPTION ARRAY-FINAL SET

3. ANALYSIS

A major portion of the study effort was devoted to collection and generation of relevant trajectory data for all phases of the mission. In order to interpret this data in terms of mission payload requirements it was also necessary to select orbits at Mars, specify characteristic velocity requirements for midcourse corrections and Mars orbit rendezvous, assume engine performance parameters, compute Mars descent and ascent performance ratios and determine hardware weights for key subsystems of the total spacecraft. The final values used to obtain quantitative study results are itemized in Table 1. A brief description of the analysis supporting these values is given below while more complete discussions are presented in Appendix A.

The Mars orbit selections are given in terms of Mars radii from the center of the planet. A conversion of these units to kms altitude is provided by Figure 9. Orbit determination accuracy, propulsion performance, rendezvous requirements, and sterilization altitude (1000 km) were the important parameters affecting selection of the various orbit sizes. A detailed discussion of the orbit selections is given in Section A.2 of the appendix.

The characteristic velocity requirements for midcourse corrections were estimated from results of previous studies and experience. While they are probably representative of the actual midcourse correction requirements, future definitive

Table 1

PAYLOAD PERFORMANCE RELATED DATA

MARS ORBITS (In Mars Radii, $R_♂$ = 3380 km)	$R_p \times R_a$
Initial Capture Orbit	2.0 x 2.0
Entry Orbit	0.9 x 2.0
Parking Orbit (Direct Launch)	1.1 x 1.1
Pre-Rendezvous Orbit (Capsule Lanuch)	1.1 x 1.3
Pre-Rendezvous Transfer Orbit (Bus)	1.3 x 2.0
Rendezvous Orbit	1.3 x 1.3

CHARACTERISTIC VELOCITY REQUIREMENTS	
Midcourse Corrections	
Earth to Mars	150 m/sec
Mars to Earth (Direct)	200
(Venus Swingby)	300
Deorbit	510
Rendezvous	740

ENGINE PERFORMANCES	Prop.	ISP	f
Earth to Mars Capture Orbit	$OF_2-B_2H_6$	400 sec	0.111
Deorbit	$N_2O_4-N_2H_4$	310	0.111
Terminal Descent	$N_2O_4-N_2H_4$	310	0.15
Launch From Mars (First Stage)	$N_2O_4-N_2H_4$	310	0.15
(Second Stage)	$N_2O_4-N_2H_4$	310	0.20
Rendezvous	$N_2O_4-N_2H_4$	310	0.111
Mars Escape	$N_2O_4-N_2H_4$	310	0.111
Mars to Earth	N_2H_4	235	-

MARS DESCENT PAYLOAD RATIOS	$W_o:W_{pl}$
Out-of-Orbit (3- 4,000 lb. Lander)	2.0:1
Out-of-Orbit (8-10,000 lb. Lander)	2.5:1
Direct Entry (8-10,000 lb. Lander)	3.0:1

MARS ASCENT PAYLOAD RATIOS	$W_o:W_{pl}$
To Parking Orbit (1.1 x 1.1)	5.75:1
To Rendezvous (1.3 x 1.3)	6.85:1

HARDWARE WEIGHTS*	
Pre-Rendezvous Capsule	440 lbs.
Rendezvous Bus	610
Ancillary Landed Equipment	250
Landing Gear	0.08 x Landed Weight
Returning Spacecraft	
Direct Return (w/o rendezvous)	580
(with rendezvous)	630
Venus Swingby (with rendezvous)	660
Earth Reentry Capsule	
w/o rendezvous	50
with rendezvous	55

* From the concurrent JPL Preliminary Design Study of the AMSR Mission.

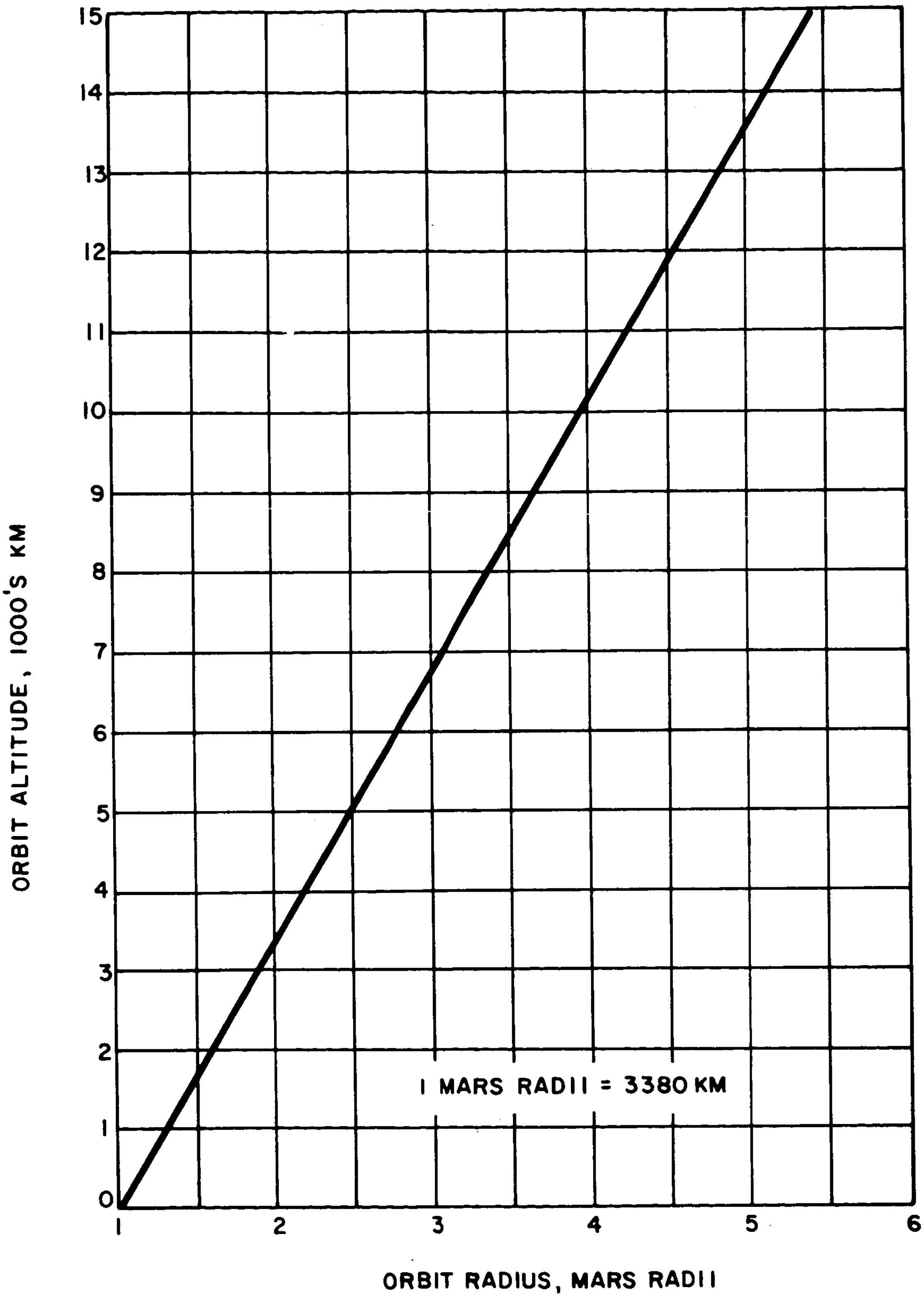

FIGURE 9. CONVERSION OF ORBIT RADIUS TO ORBIT ALTITUDE

studies of the AMSR mission should include a specific guidance and control analysis. The velocity requirement for Mars orbit rendezvous is the result of a proposed rendezvous scheme which is developed in Section A.6 of the appendix.

Propellants were selected and specific impulses (ISP) and propulsion structure factors (f) assumed for each powered maneuver of the mission. For impulsive maneuvers (all maneuvers except Mars terminal descent and ascent) the resulting mass fraction was computed by the equation

$$\frac{m_f}{m_o} = 1 - (1 + f)(1 - e^{-\frac{dV}{gISP}}) ,$$

where m_f is the final mass (exclusive of tanks, engines, and related structure used in the maneuvers), m_o is the total initial mass, and dV is the required characteristic velocity change. Sterilization was assumed for all the propellants except OF_2-B_2H_6 (used for the Mars orbit capture maneuver). No structure factor is given for the Mars to Earth midcourse correction maneuvers (N_2H_4) since that propulsion system has been included in the total returning spacecraft weight estimates. A recent re-examination (Kelley 1967) of the assumed ISP's and f's indicated that the values of f shown in Table 1 tend to be optimistic for 1975 mission application.

A brief evaluation of the Mars entry phase has already been presented in Section 2.5. A more complete discussion of the parametric analysis of combined atmospheric braking and terminal powered descent done in this study is provided in Appendix A.4. As a result of that analysis and review of Voyager entry results provided by JPL, three specific descent ratios of entry weight (W_o) to landed weight (W_{Pl}) were selected for payload calculations of the three possible descent/ascent combinations shown in Figure 8.

For the out-of-orbit-descent/ascent-to-rendezvous combination the landed weight requirement was estimated between 3-4,000 lb. This permitted the first descent ratio selection to be made from Voyager descent performance results (JPL 1966b) for the same weight range. A value of 2:1 was selected as representative performance for this descent/ascent combination.

For the combination of out-of-orbit-descent/ascent-to-parking-orbit (and subsequent escape) the landed weight requirement was estimated between 8-10,000 lb. This weight range is outside the current Voyager requirements so the descent ratio selection depended upon the results of preliminary descent analysis performed in this study. From these results a descent ratio of 2.5:1 was chosen as representative performance for this descent/ascent combination. To provide an appreciation for this choice, ratios of entry weight to landed weight have been plotted in Figure 10 as a function of ballistic coefficient for several ignition altitudes (h). The entry trajectory data used in preparing the figure was provided by JPL from the same computer program (Feitis 1966) used in their Voyager studies. The data was generated with the JPL VM-8 (5 mb) Mars model

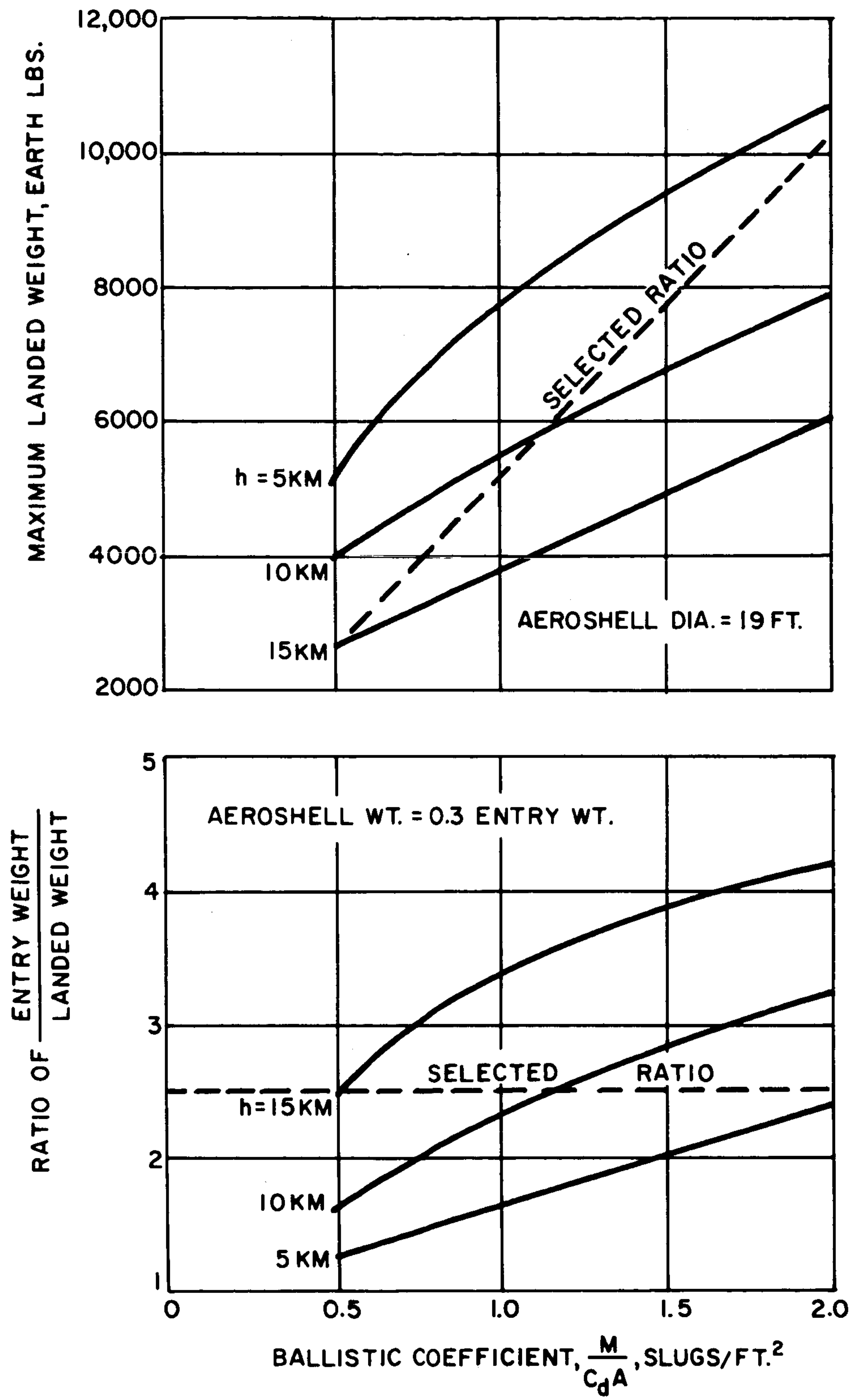

FIGURE 10. OUT-OF-ORBIT DESCENT PERFORMANCE

(JPL VM-8, 5mb MODEL ATMOSPHERE)

atmosphere. The specified entry conditions are: velocity = 3.80 km/sec, path angle = -15°, entry altitude = 800,000 ft. Characteristic velocity requirements were computed for terminal powered descent from the gravity-turn approximation due to Schleuter (1965). The weight ratios are based on a staged aeroshell weight equal to 30 percent of the entry weight and on the engine performance assumptions for terminal descent given in Table 1.

From Figure 10 the ratio of 2.5:1 would appear to be conservative. Lower ratios are possible while still providing sufficient altitude (5 km or 16,400 ft) for the terminal powered descent. However, due to the preliminary nature of the analysis some contingency was felt to be necessary. In particular, the aeroshell weight and structure factor assumptions have not been justified by design considerations. Also, the response time between aeroshell staging and full terminal thrust and the deceleration limit of a specific capsule design are unknown.

Also plotted on Figure 10 is the maximum landed weight as a function of ballistic coefficient again for several ignition altitudes. This landed weight was computed assuming a drag coefficient, C_d = 1.4 and a fixed aeroshell diameter of 19 ft (maximum for the standard 20 foot Saturn V shroud). For an assumed ratio of 2.5:1 the dotted line shows the variation in maximum landed weight. For landed weights between 8-10,000 lb the required ballistic coefficient is between 1.5-2.0 and the ignition altitude between 5-10 km.

The combination of direct-entry-descent/ascent-to-parking-orbit was also estimated to have a landed weight requirement between 8-10,000 lb. The choice of descent ratio for this combination was therefore also determined on the basis of the study descent analysis. A ratio of 3:1 was chosen as representative. This selection is supported by results similar to Figure 10 plotted in Figure 11. The only difference is in the entry conditions, for which: velocity = 5.75 km/sec, path angle = -20°, entry altitude = 800,000 ft.

The selected ratio of 3:1 is perhaps not nearly as conservative as that for out-of-orbit descent. Furthermore, for a fixed aeroshell of diameter = 19 ft the 8-10,000 lb range in landed weight is not possible for ignition altitudes above 3 km (10,000 ft). A maximum landed weight of 8500 lb is available for a ratio of 3:1 (ballistic coefficient = 2.0). The obvious solution for more landed weight would be to increase the diameter of the aeroshell. However, this would require some alteration to the Saturn V shroud and almost certainly further complicate the stability and control problems of an already large aeroshell. Another possible solution would be to incorporate lift in the atmospheric braking portion of descent. No solution to this problem is recommended in this study, since much more detailed analysis would be required to make such a decision. In fact, a detailed analysis of the entire entry problem is recommended for landed weight requirements in the range of 8-10,000 lb.

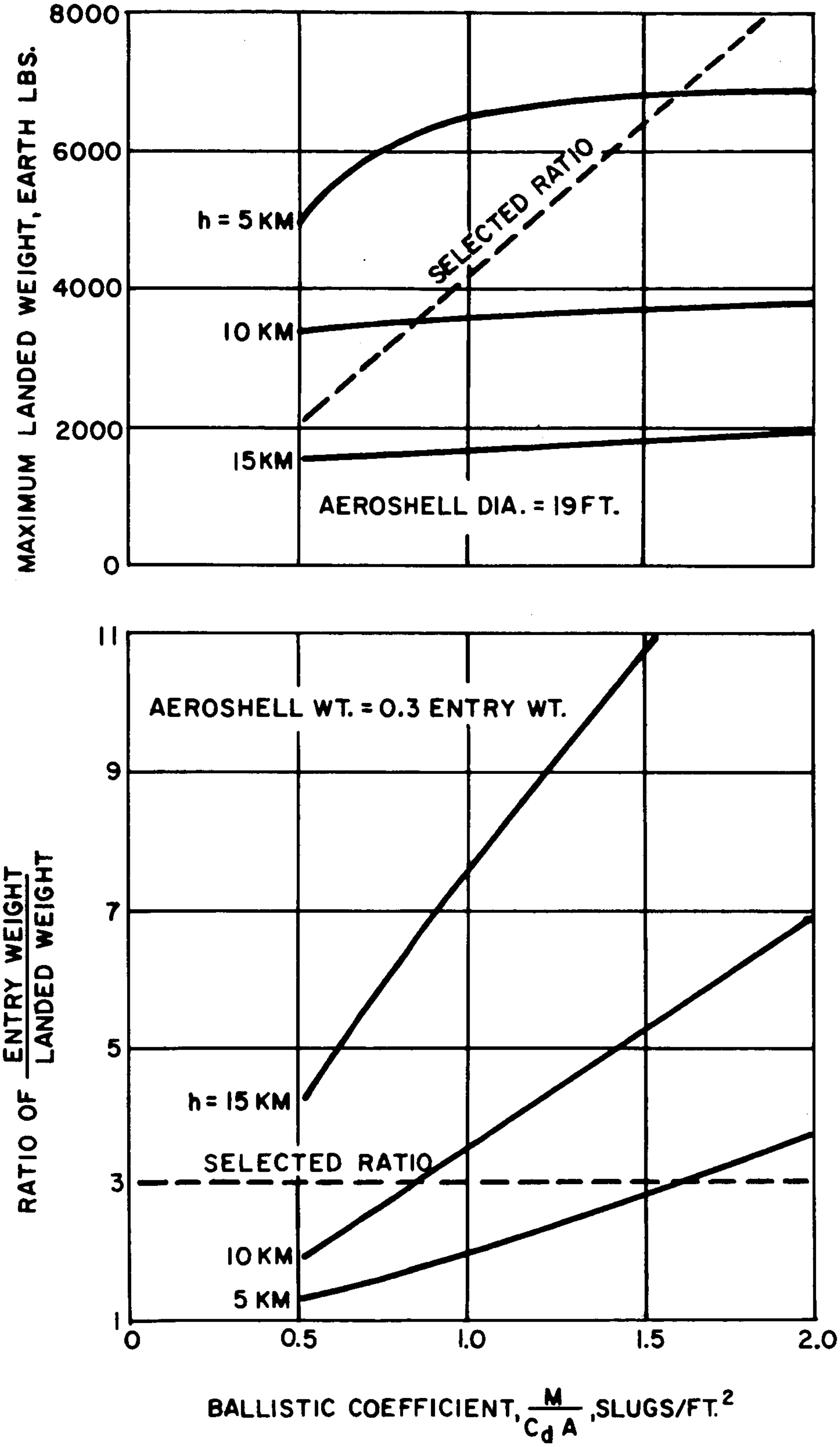

FIGURE 11. DIRECT ENTRY DESCENT PERFORMANCE
(JPL VM-8, 5mb MODEL ATMOSPHERE)

The Mars ascent payload ratios were determined by optimized, staged, ascent trajectory analysis (Teren and Spurlock 1966) contributed to the study by Lewis Research Center, NASA for the ascent engine performance assumptions given in Table 1. The ratios given in Table 1 are expressed as lift-off weight (W_o) to payload weight injected into orbit (W_{Pl}). A more complete discussion of these results is presented in Section A.5 of the appendix.

The hardware weights given in Table 1 were supplied by the concurrent preliminary study of systems requirements for the AMSR mission conducted at JPL (1967). These weights are the basis for the subsequent weight statements presented in the report. They are based upon a number of unverified operational feasibility assumptions and should not be construed as final systems weights. Rather, they were used in the study as a tool in obtaining representative weights for the various mission modes considered.

The Mars orbit selections, engine performance assumptions, characteristic velocity requirements, and returning spacecraft weights were all combined with interplanetary trajectory data to formulate plots of transfer and Mars orbit payloads as a function of Mars arrival and departure dates. Figure 12 is a plot of payload weight in a circular 2 Mars radii orbit versus Mars arrival dates between opposition and conjunction. The Earth escape and Mars approach velocities have been simultaneously optimized to maximize the payload in orbit for flight

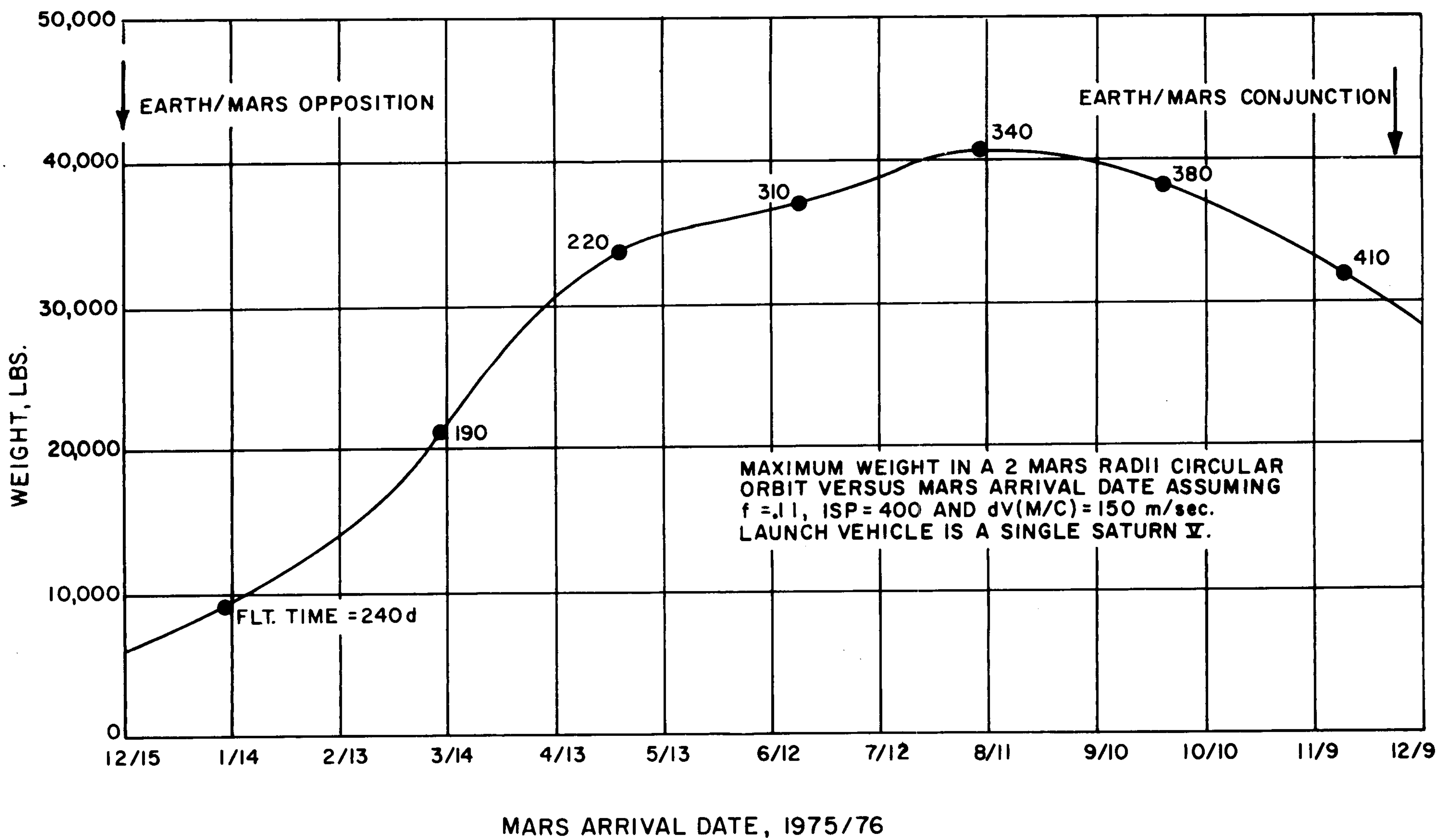

FIGURE 12. MAXIMUM MARS ORBIT PAYLOAD CAPABILITY

times of less than one year (with slightly longer trip times at conjunction to maintain continuity). Figure 13 is a plot of maximum interplanetary payload as a function of Mars arrival date for use with Mars direct entry descent option. Figures 12 and 13 do not use the same trajectory data and should not be used interchangeably.

Figure 14 is a plot of required payload in orbit to inject the prescribed returning spacecraft on Mars/Earth transfers versus Mars departure date for short stay times. Orbit payloads for both direct and Venus swingby return options are plotted. The direct return curve applies for both ascent to rendezvous and ascent to parking orbit options. The difference in returning spacecraft weight (630 vs. 580 lb) is offset by the different parking orbits used (1.3 vs. 1.1 Mars radii). Also shown are the corresponding Earth reentry speeds and the Venus miss distance (for the Venus swingby option) as a function of Mars departure date. Figure 15 is similar to Figure 14 but pertains to long Mars stay time and only direct returns are applicable.

The payload feasibility of a mission mode is determined by beginning with the required payload in Mars orbit (Figure 14 or 15 depending upon stay time) for a selected Mars departure date. Working backward the payload requirements are determined for rendezvous (if applicable), Mars launch, and descent phases. The appropriate weights are combined to determine the total <u>required</u> weight injected into Mars orbit or directly entering

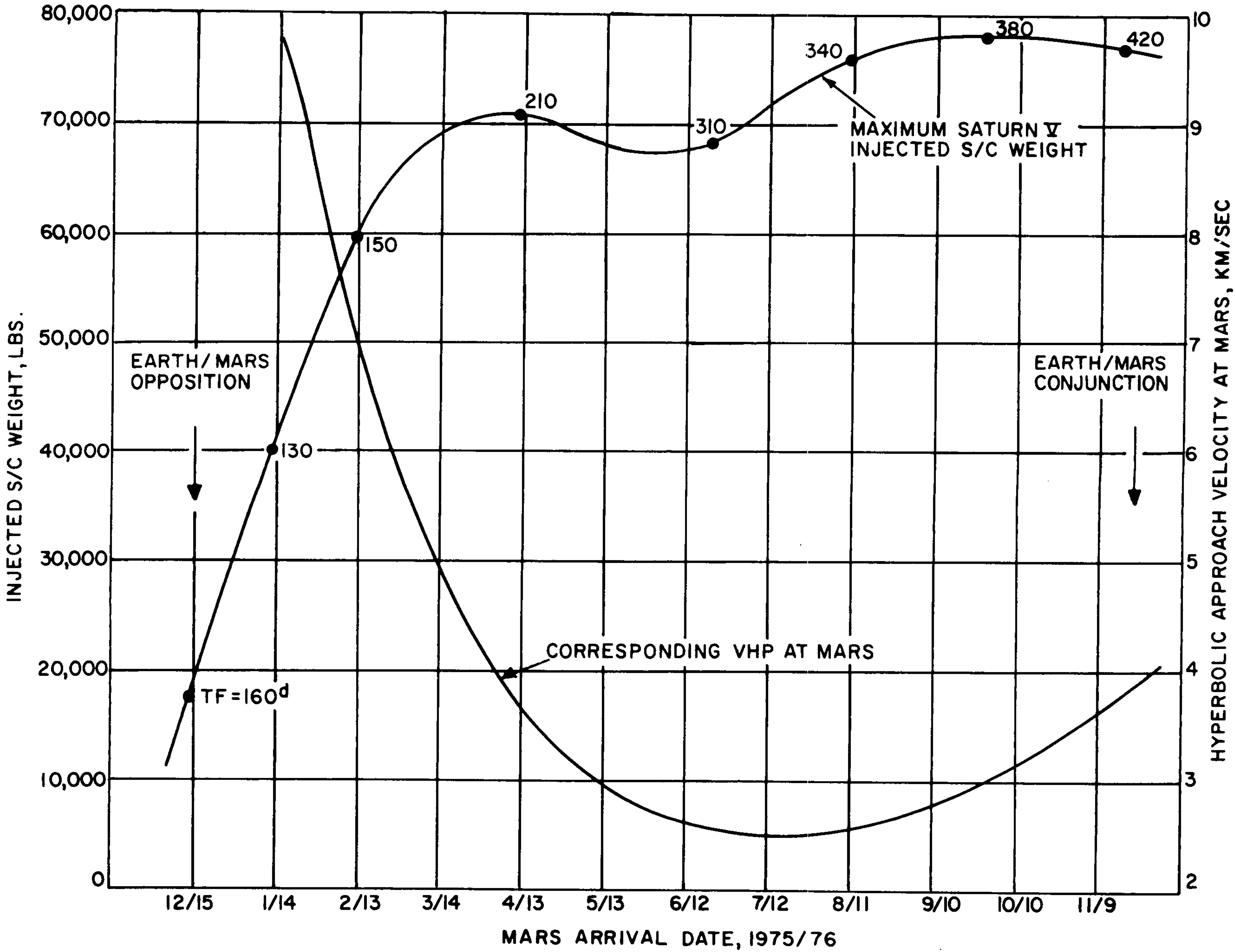

FIGURE 13. MAXIMUM OUTBOUND TRANSFER PAYLOAD

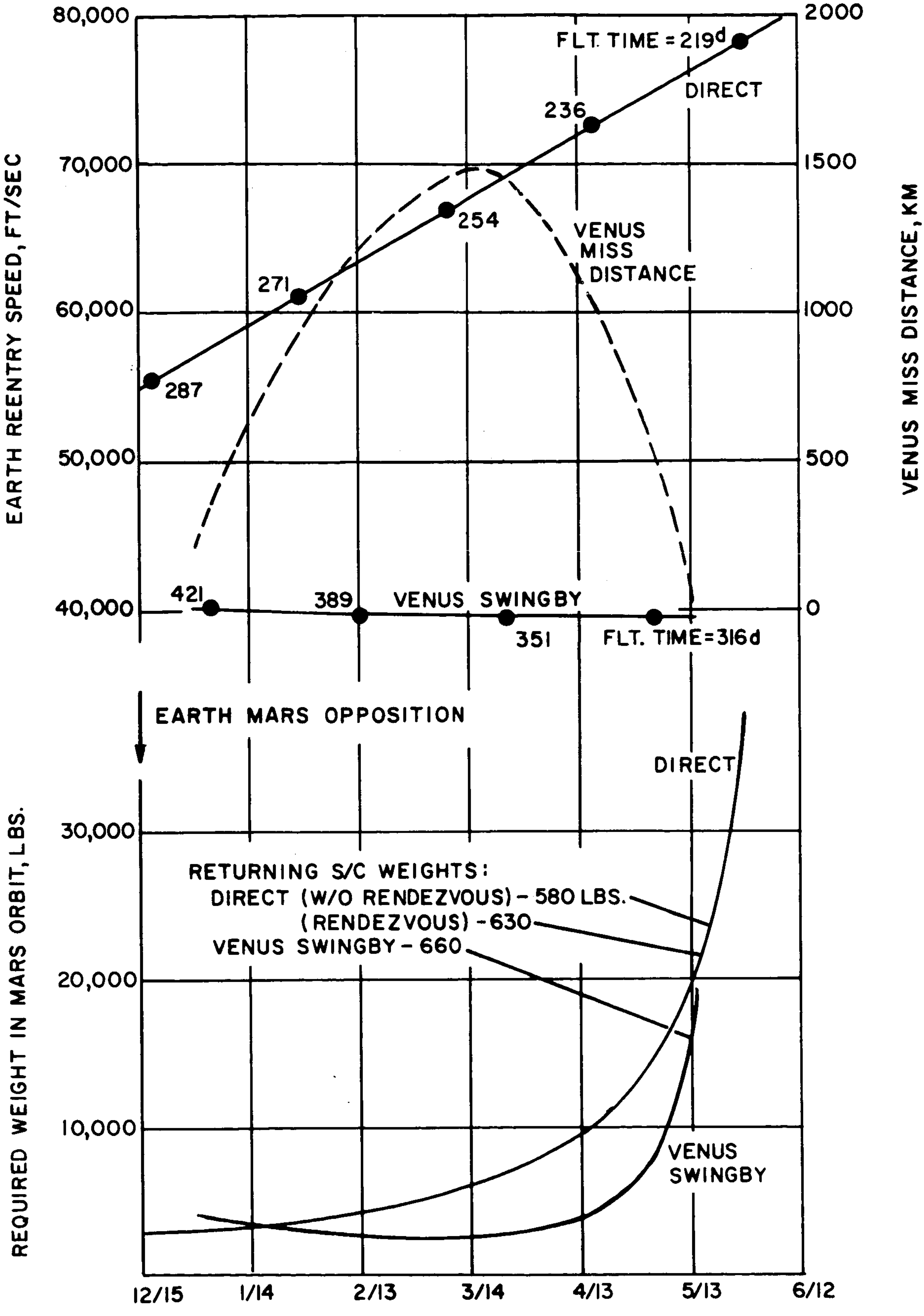

FIGURE 14. REQUIRED ORBIT PAYLOAD FOR RETURN TRANSFER, SHORT STAY TIME

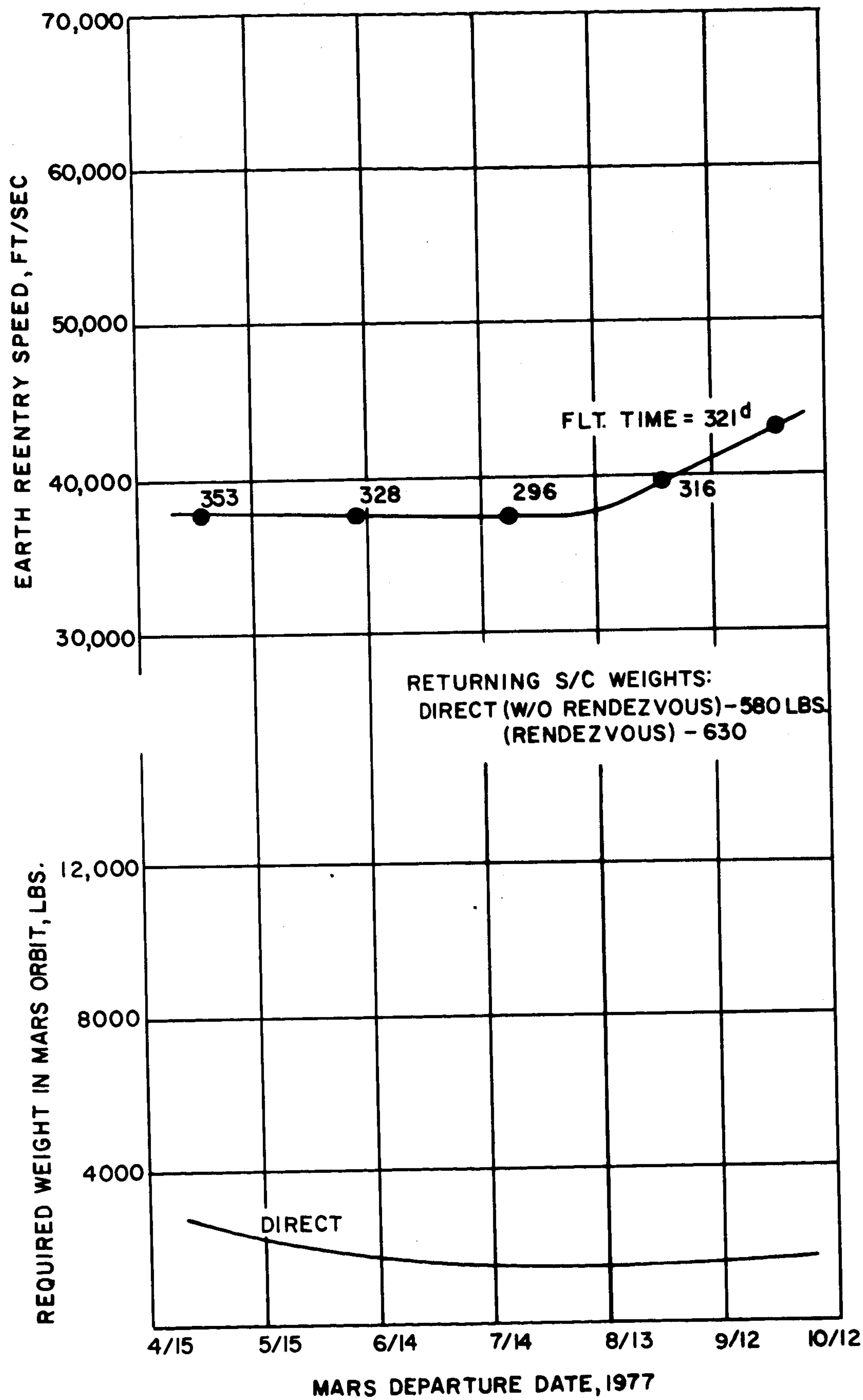

FIGURE 15. REQUIRED ORBIT PAYLOAD FOR RETURN TRANSFERS, LONG STAY TIME

the Martian atmosphere. This weight is then compared with the maximum available weight (Figure 12 or 13) at a Mars arrival date consistent with the stay time selected. The procedure is repeated if necessary until a combination of Mars arrival date and stay time is found with a required total payload less than the maximum payload available for that arrival date. If this condition cannot be satisfied then that particular mission mode is not feasible under the assumptions being used.

4. MODE DESCRIPTIONS

Of the 12 mission modes which can be formulated from Figure 8, four were found to require a total weight within the capability of a single Saturn V launch vehicle with the imposed study constraints and assumptions. Each of these mission modes is described below. A fifth mode which cannot meet the Earth reentry speed constraint of 45,000 ft/sec within the capability of a single Saturn V is also presented in the event that a much higher entry speed of about 70,000 ft/sec might be feasible.

4.1 Mode 1

The option selections for this configuration are shown in Figure 16. The mode is characterized by near-minimum energy outbound and return transfers and long stay time. Rendezvous is not used. The transfer trajectory data used for payload calculations are as follows:

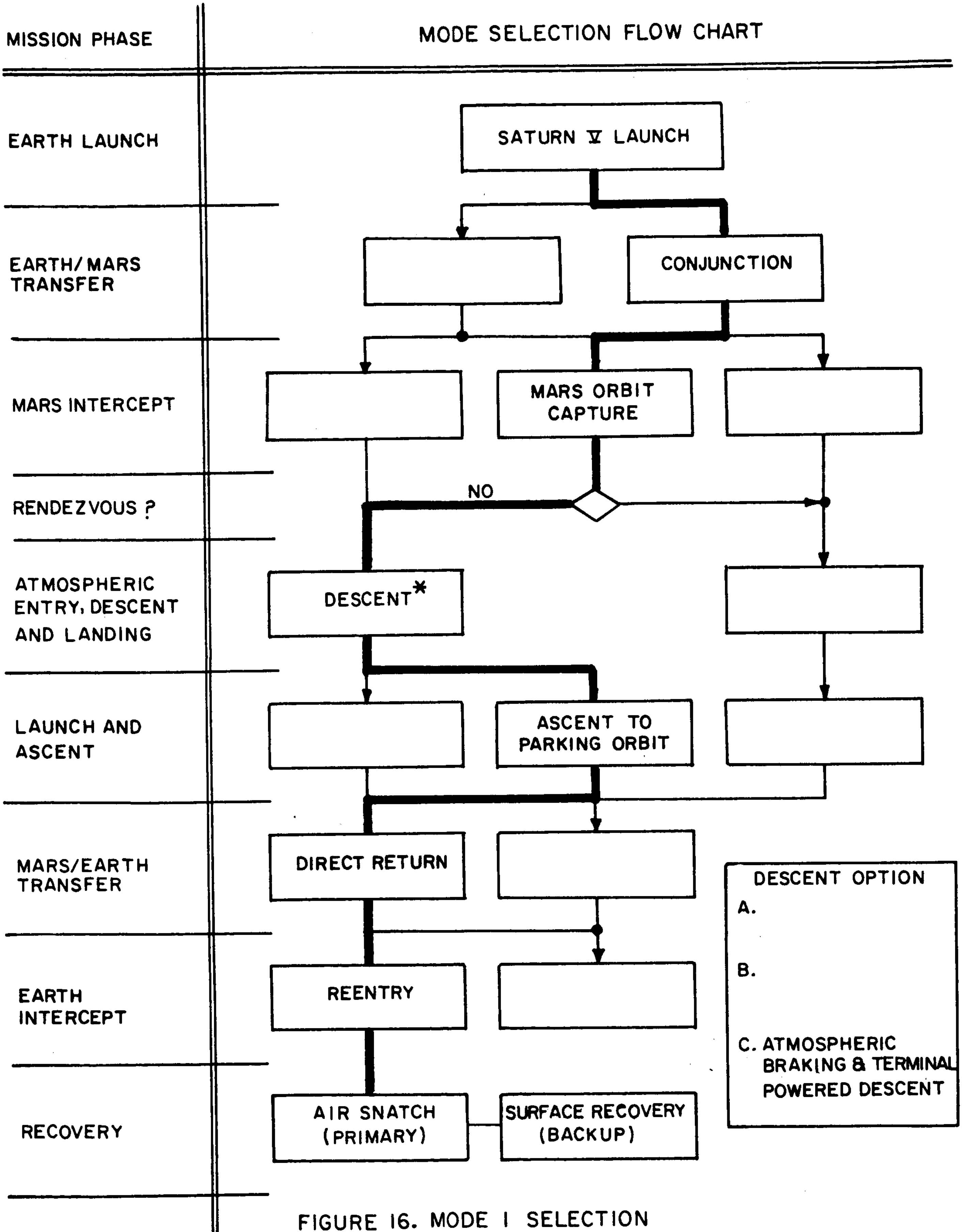

FIGURE 16. MODE I SELECTION

Earth launch date	- Sept. 9, 1975
Outbound flight time	- 364 days
Mars arrival date	- Sept. 7, 1976
Mars hyperbolic approach velocity	- 2.72 km/sec
Stay time	- 306 days
Mars hyperbolic departure velocity	- 2.98 km/sec
Mars departure date	- July 10, 1977
Return flight time	- 305 days
Earth arrival date	- May 11, 1978
Reentry speed	- 37,650 ft/sec
Total trip time	- 975 days

The weight breakdown determined from payload calculations is as follows:

Saturn V capability (Figure A1)	- 75,000 lb
Earth/Mars spacecraft	- 50,600
Mars orbit weight	- 26,900
Mars descent capsule	- 22,300
Mars landed weight	- 8,920
Mars launch vehicle	- 7,960
Orbit weight	- 1,380
Mars/Earth spacecraft	- 580
Earth reentry capsule	- 50

The Saturn V injected spacecraft weight is 50,600 lb. A Type II near-minimum energy transfer trajectory is used to reach Mars in one year. At least one and probably two midcourse maneuvers are executed during the transfer. The propellant for these maneuvers is cryogenic OF_2-B_2H_6, ISP = 400 sec. The assumed midcourse dV is 150 m/sec.

The spacecraft is inserted into a circular 2 Mars radii orbit at Mars arrival. The captured payload weight is 26,900 lb.

The propellant used is cryogenic OF_2-B_2H_6, ISP = 400 sec. dV is 2 km/sec.

At a predetermined point on the circular orbit the entry sequence begins with a retro maneuver dV of 550 m/sec placing the descent capsule on an entry ellipse with an atmospheric entry angle (γ) of about -15°. Using a descent weight ratio of 2.5:1 the landed weight is 8920 lb. The propellant used for orbit retro and terminal descent is sterilized liquid storable N_2O_4-N_2H_4, ISP = 310 sec.

The sample is collected and stored in an Earth reentry capsule on the Mars launch vehicle. The vehicle weighs 7,960 lb. Landing gear and ancillary equipment for descent and sample collection totaling 960 lb are left on the Mars surface. The launch vehicle consists of three stages. Two stages are used to reach a circular parking orbit of 1.1 Mars radii. A launch to orbit payload ratio of 5.75:1 yields an orbit weight, including the third stage, of 1,380 lb. The first stage characteristic velocity is 10,400 ft/sec, the second stage 3,000 ft/sec.

A longer stay time in Mars orbit must be made either before or after descent to the surface until the Mars/Earth phasing is compatible with a low energy return transfer. When the acceptable departure date finally arrives the third stage injects a 580 lb spacecraft onto a hyperbolic escape trajectory from Mars. The launch vehicle propellant is liquid storable N_2O_4-N_2H_4, ISP = 310 sec. The escape dV is 2.25 km/sec.

The return transfer flight time is 305 days. Two midcourse maneuvers are possible and share a total midcourse capability of 200 m/sec. Midcourse propulsion uses monopropellant N_2H_4, ISP = 235 sec. Upon approach to Earth, the Earth reentry container, which weighs 50 lb is ejected from the spacecraft and reenters the Earth's atmosphere at 37,650 ft/sec. Following aerodynamic braking the capsule makes a terminal parachute descent during which an air snatch is attempted. A backup surface recovery is initiated if the air snatch is missed.

The Mars arrival date is Sept. 7, 1976. Arrival dates at Mars during 1976 which have maximum orbit payloads in excess of 26,900 lb extend from about April 1, 1976 to the end of the year (see Figure 12). To shorten the stay time at Mars it would seem logical to select a later arrival date. However, the Mars/Earth trip time increases with later arrival dates erasing the effect of a shorter Mars stay time on total flight time. The total flight time is 975 days.

The Saturn V payload capability for this mission mode is about 75,000 lb. Compared with the required total spacecraft weight of 50,000 lb almost 25,000 lb (50 percent) contingency is available. However, a number of important assumptions used to compute the required payload weight have not been verified by design and system analysis. These assumptions include, (1) a 2.5:1 descent performance ratio to land almost 9,000 lb on Mars, (2) small injection errors by an unmanned Mars ascent vehicle from a random surface, (3) structure factors (f) for all propulsion stages, and (4) Earth reentry and recovery of a 1-2 lb sample using a 50 lb entry capsule.

4.2 Mode 2

The option selections for this mode are shown in Figure 17. The mode is similar to Mode 1 except entry in the Martian atmosphere is directly from hyperbolic approach, instead of via orbit capture. The transfer trajectory data used for payload calculations is identical to Mode 1. The weight breakdown determined from payload calculations is as follows:

Saturn V capability	- 75,000 lb
Earth/Mars spacecraft	- 28,850
Mars descent capsule	- 26,750
Mars landed weight	- 8,920
Mars launch vehicle	- 7,960
Orbit weight	- 1,380
Mars/Earth spacecraft	- 580
Earth reentry capsule	- 50

The Saturn V injected spacecraft weight is 28,850 lb. A Type II near-minimum energy transfer trajectory is used to reach Mars in one year. Two midcourse maneuvers are executed during transfer sharing a 150 m/sec capability. Midcourse propulsion uses monopropellant N_2H_4, ISP = 235 sec.

The midcourse propulsion hardware is jettisoned at Mars arrival and a 26,750 lb capsule enters the Martian atmosphere directly. There is no orbit capture. Using an overall descent weight ratio of 3:1 the landed weight is 8,900 lb. The

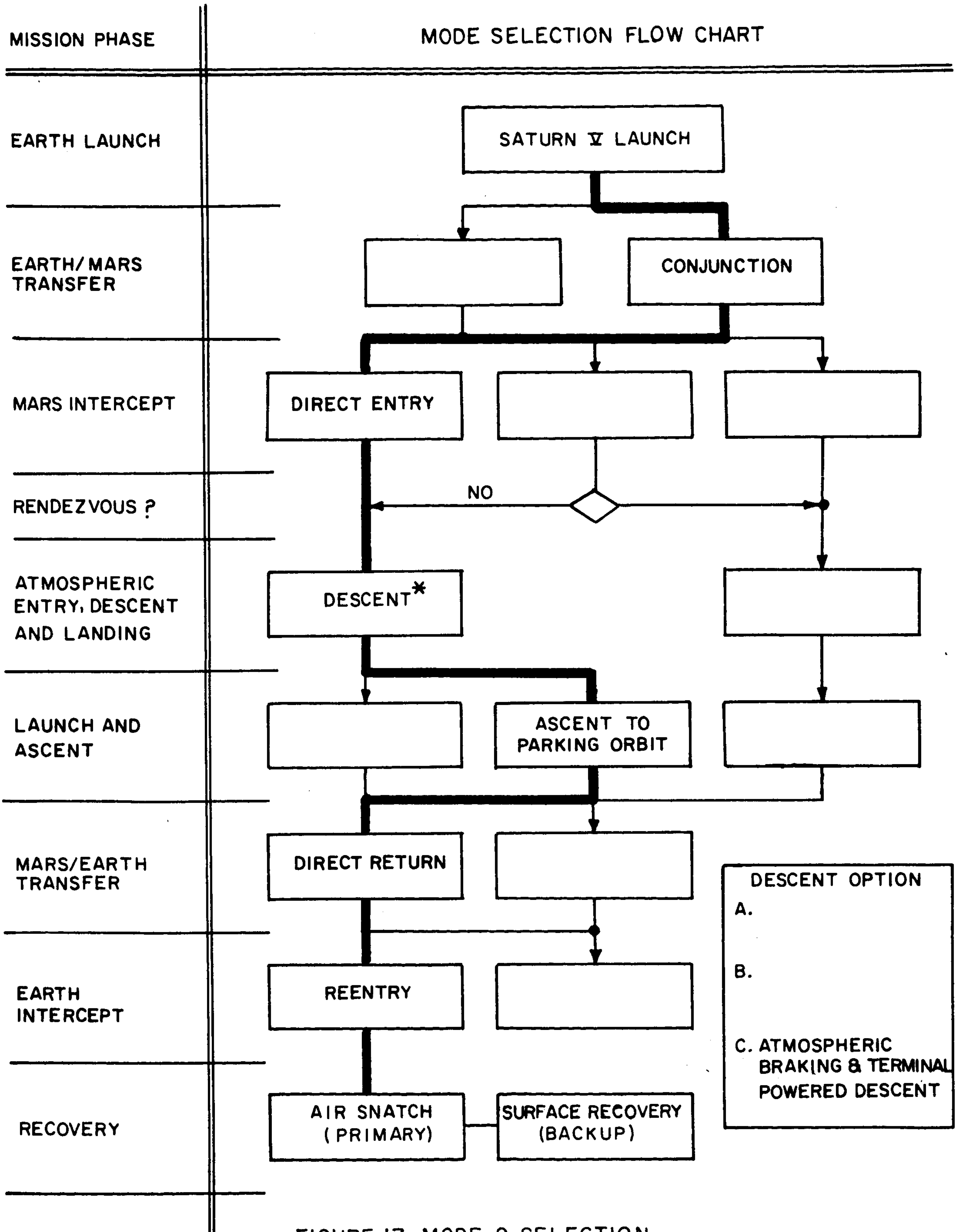

FIGURE 17. MODE 2 SELECTION

propellant used for terminal descent is sterilized liquid storable N_2O_4-N_2H_4, ISP = 310 sec.

Mars launch, Mars/Earth transfer and Earth intercept phases are identical to Mode 1. The total flight time is 975 days. The Saturn V payload capability for this mission mode is about 75,000 lb. While this reflects a payload contingency of more than 45,000 lb (150 percent) above the 28,850 lb required spacecraft weight, the question of feasibility does not center around payload, but systems and design requirements. These requirements involve (1) extremely accurate guidance and control requirements for direct shallow Mars entry (-20°) from a hyperbolic approach, and (2) aeroshell design incorporating lift, or aeroshell diameters in excess of the 19 foot limit of the standard Saturn V shroud, or another solution which can land about 9,000 lb on Mars (see Figure 11). In addition to these requirements the same reservations on assumptions noted for Mode 1 also apply here.

4.3 Mode 3

The option selections for this mode are shown in Figure 18. The mode is characterized by the use of Mars orbit rendezvous. Outbound and return transfer trajectories and stay time are identical to Modes 1 and 2. The weight breakdown determined from payload calculations is as follows:

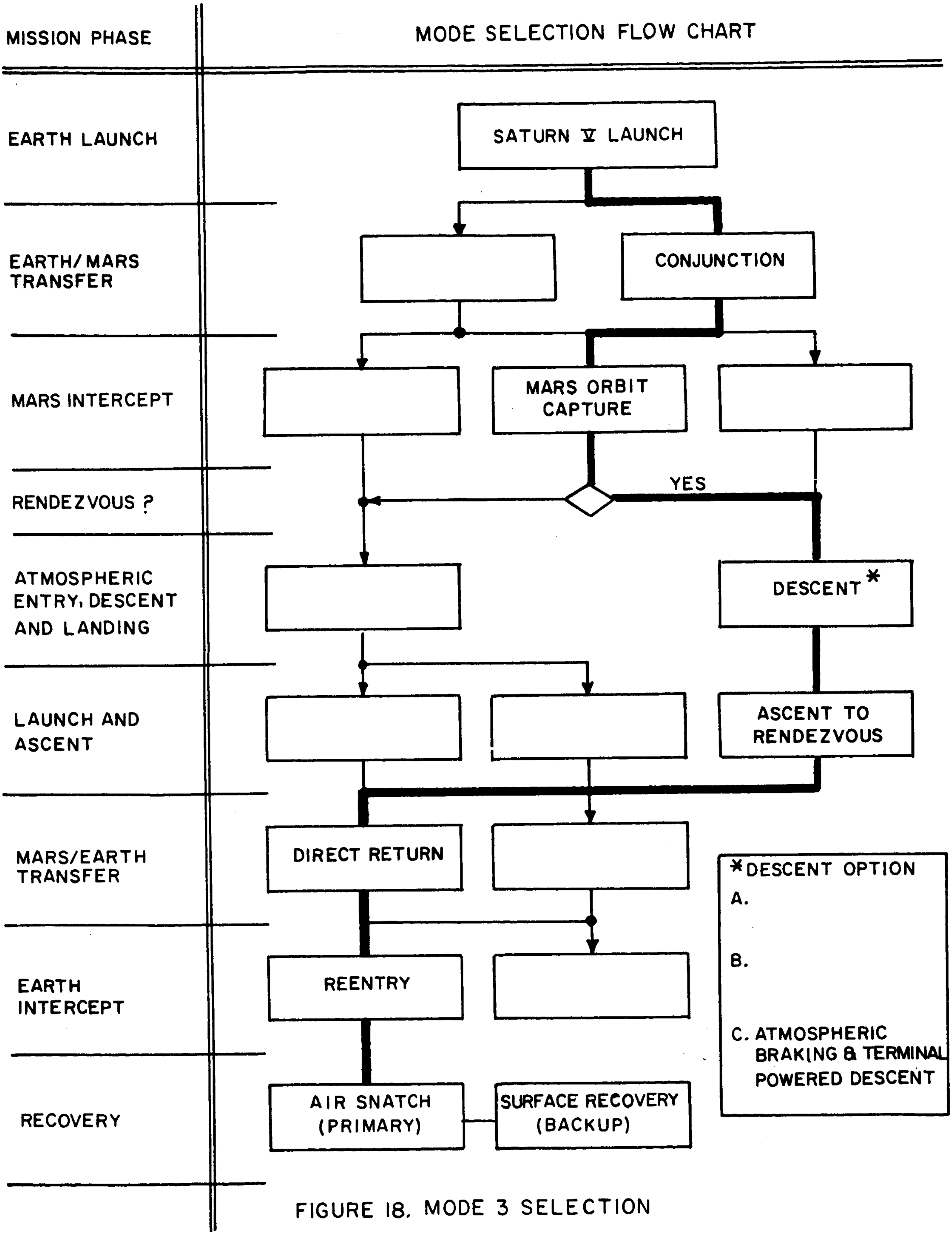

FIGURE 18. MODE 3 SELECTION

Saturn V capability	- 75,000 lb
Earth/Mars spacecraft	- 21,200
Mars orbit weight	- 11,250
Mars rendezvous bus	- 2,700
Mars descent capsule	- 7,100
Mars landed weight	- 3,550
Mars launch vehicle	- 3,020
Mars rendezvous capsule	- 440
Orbit trimmed weight	- 1,500
Mars/Earth spacecraft	- 630
Earth reentry capsule	- 55

The launch, Earth/Mars transfer and Mars intercept phases are identical to Mode 1. The orbit payload weight after Mars capture is 11,250 lb. Midcourse and capture propulsion uses cryogenic OF_2-B_2H_6 propellant, ISP = 400 sec. The capture dV is 2 km/sec.

The operations at Mars include rendezvous as opposed to launch to escape used in Modes 1 and 2. Hence only a portion of the 11,250 lb orbit weight is used in the descent capsule (7,100 lb). A remaining orbit payload of 2,700 lb makes up the rendezvous hardware, the Mars/Earth transfer vehicle (excluding capsule), and Mars escape propulsion.

A retro maneuver of 550 m/sec places the descent capsule on an entry ellipse (γ = -15°). Using a descent weight ratio of 2:1 the landed weight is 3,550 lb. The propellant used for orbit retro and terminal descent is sterilized liquid storable N_2O_4-N_2H_4, ISP = 310 sec.

A Mars surface sample is collected and stored in the 55 lb Earth reentry capsule on the Mars launch vehicle. The launch vehicle consists of two stages and weighs 3,020 lb. Landing gear and ancillary equipment for descent and sample collection weighing 530 lb are left on the Mars surface.

Injection into a 1.1 x 1.3 transfer orbit occurs after the first burn of the second stage. A second burn of the second stage is made at apoapsis of the transfer ellipse to place the ascent (rendezvous) capsule in a circular 1.3 Mars radii orbit. A launch to final orbit weight ratio of 6.85:1 yields a rendezvous capsule weight (including the Earth reentry capsule) of 440 lb. The launch vehicle propellant is liquid storable N_2O_4-N_2H_4, ISP = 310 sec. The first stage characteristic velocity is 10,200 ft/sec; for the second stage it is 4,500 ft/sec.

Following injection, the ascent capsule and the rendezvous bus (still in a circular 2.0 Mars radii orbit) are tracked to determine an accurate set of orbit elements. The rendezvous bus initiates rendezvous with a retro maneuver of 280 m/sec to transfer to a 2 x 1.3 Mars radii ellipse. A second maneuver of 315 m/sec circularizes the bus at 1.3 Mars radii and simultaneously initiates the terminal phase of rendezvous and docking. Additional dV's of 90 m/sec for out-of-plane error (2° in inclination) and 55 m/sec for final approach and docking bring the total required rendezvous dV to 740 m/sec. This is all performed by the rendezvous bus using liquid storable

N_2O_4-N_2H_4 propellant, ISP = 310 sec. The ascent capsule maintains radar and communication links with the bus and responds to attitude alignment commands from the bus but performs no dV maneuvers after circular orbit insertion.

After docking the return transfer vehicle is trimmed to 1,500 lb including escape propulsion. The same Mars/Earth transfer as used in Modes 1 and 2 is initiated when the planetary phasing is correct. The injected return spacecraft weighs 630 lb. Escape propulsion uses liquid storable N_2O_4-N_2H_4, ISP = 310 sec. Escape dV is 2.23 km/sec. The total flight time is 975 days.

The Saturn V payload capability for this mission mode is 75,000 lb. Compared with the required total spacecraft weight of 21,200 lb a payload contingency of more than 50,000 lb (250 percent) exists. In fact, from a weight standpoint, if two complete Mode 3 payloads were launched there would still be more than a 16,000 lb (75 percent) contingency for each payload. Feasibility of this mode centers on the design and development of an automatic Earth-controlled rendezvous and docking system at Mars, as well as verification of structure factor and launch vehicle accuracy assumptions which affect the total payload requirement and Earth reentry capsule design for entry speeds up to 40,000 ft/sec.

4.4 Mode 4

The option selections for this mode are shown in Figure 19. The mode is characterized by a short stay time, use

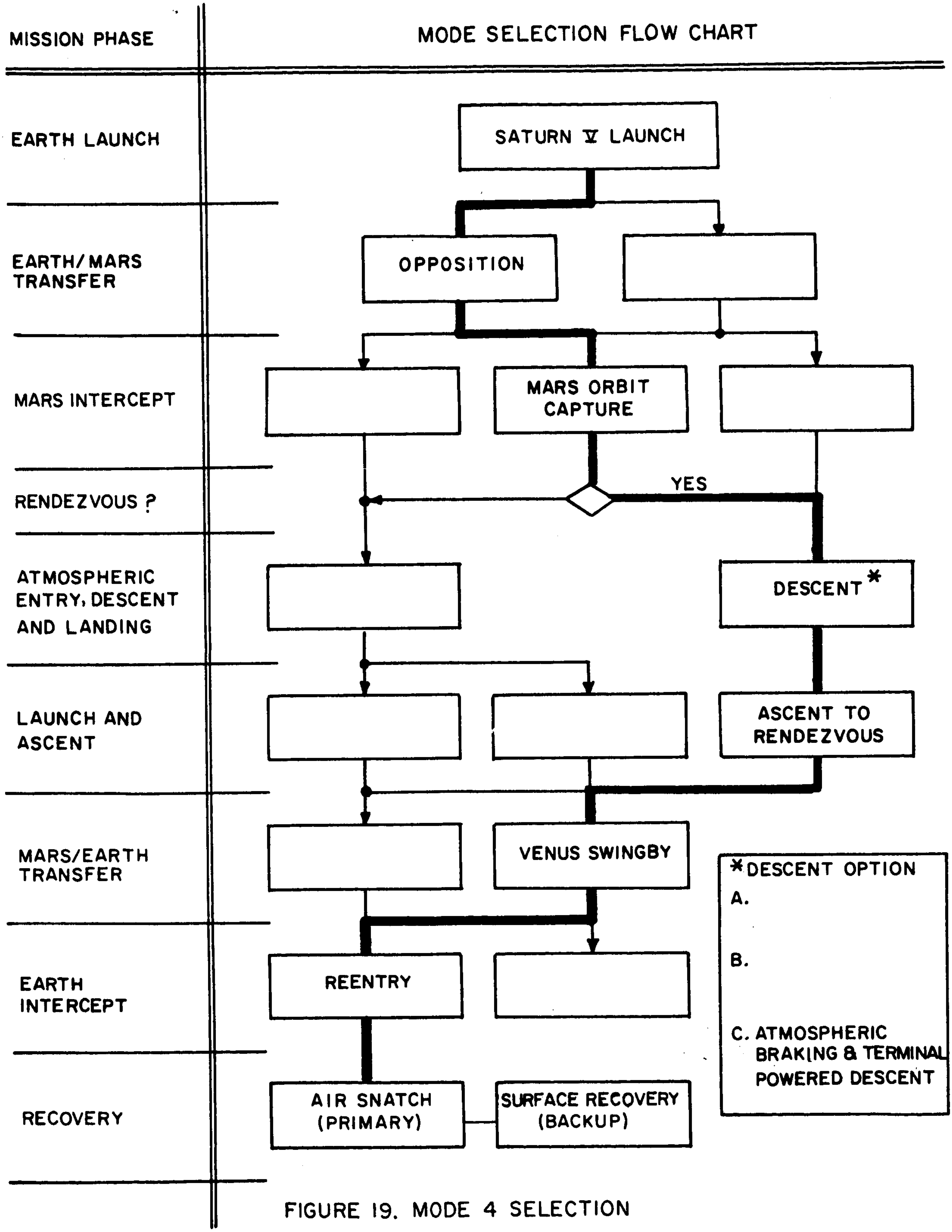

FIGURE 19. MODE 4 SELECTION

of Mars orbit rendezvous and a Venus swingby on the Mars/Earth return transfer. The transfer trajectory data used for payload calculations are as follows:

Earth launch date	- Sept. 14, 1975
Outbound flight time	- 200 days
Mars arrival date	- April 1, 1976
Mars hyperbolic approach velocity	- 4.05 km/sec
Stay time	- 12 days
Mars hyperbolic departure velocity	- 5.81 km/sec
Mars departure date	- April 13, 1976
Return flight time	- 337 days
Venus miss distance	- 1134 km
Earth arrival date	- March 16, 1977
Reentry speed	- 39,600 ft/sec
Total trip time	- 549 days

The weight breakdown determined from payload calculations is as follows:

Saturn V capability	- 69,000 lb
Earth/Mars spacecraft	- 36,150
Mars orbit weight	- 14,450
Mars rendezvous bus	- 5,900
Mars descent capsule	- 7,100
Mars landed weight	- 3,550
Mars launch vehicle	- 3,020
Mars rendezvous capsule	- 400
Orbit trimmed weights	- 3,900
Mars/Earth spacecraft	- 660
Earth reentry container	- 55

The Type I Earth/Mars trajectory used for payload calculation has a trip time of 200 days with a Mars arrival on April 1, 1976. The Saturn V injected spacecraft weight is

36,150 lb. Two midcourse maneuvers are possible during the outbound transfer. The spacecraft is inserted into a circular 2 Mars radii orbit at Mars arrival. The payload weight following orbit capture is 14,450 lb. The propulsion for both midcourse and orbit capture maneuvers is cryogenic OF_2-B_2H_6, ISP = 400 sec. Total dV is 3.05 km/sec.

The subsequent mission phases through rendezvous and docking are identical to Mode 3. However, because a higher energy Mars/Earth transfer follows, the rendezvous bus weight including escape propulsion is increased to 5,900 lb. Following docking the trimmed vehicle weighs 3,900 lb.

The return transfer is initiated just 12 days after Mars arrival. The injected spacecraft weighs 660 lb. The escape propulsion uses liquid storable N_2O_4-N_2H_4 propellant, ISP = 310 sec. Escape dV is 4.2 km/sec.

A Venus swingby is included in the return transfer to reduce the Earth reentry speed to 39,600 ft/sec. (Entry speeds for a direct return would be greater than 50,000 ft/sec.) The Venus miss distance altitude is 1,130 kilometers (0.22 Venus radii). Four midcourse maneuvers are possible during the return transfer, with a total dV capability of 300 m/sec. Midcourse propulsion uses monopropellant N_2H_4, ISP = 235 sec. The return flight time is 337 days. Total trip time is 549 days. The Earth reentry phase is similar to that used in Modes 1-3.

Saturn V payload capability for this mission mode is about 69,000 lb. The required total spacecraft weight of

36,150 lb leaves a payload contingency of more than 32,000 lb (90 percent). Added to the feasibility reservations summarized for Mode 3, which also apply to this mode, is the increased guidance and control requirement to make the necessary Venus swingby on the return transfer.

4.5 Mode 5

The option selections of this mode are shown in Figure 20. The mode is characterized by short Mars stay time and use of Mars orbit rendezvous. Direct trajectories are used for both outbound and return transfers. The trajectory data used for payload calculations are as follows:

Earth launch date	- June 6, 1975
Outbound flight time	- 250 days
Mars arrival date	- Feb. 21, 1976
Mars hyperbolic approach velocity	- 3.52 km/sec
Stay time	- 2 days
Mars hyperbolic departure velocity	- 6.38 km/sec
Mars departure date	- Feb. 23, 1976
Return flight time	- 260 days
Earth arrival date	- Nov. 9, 1976
Reentry speed	- 64,800 ft/sec
Total trip time	- 512 days

The weight breakdown determined from payload calculations is as follows:

Saturn V capability	- 34,250 lb
Earth/Mars spacecraft	- 34,250
Mars orbit weight	- 15,625
Mars rendezvous bus	- 7,075
Mars descent capsule	- 7,100

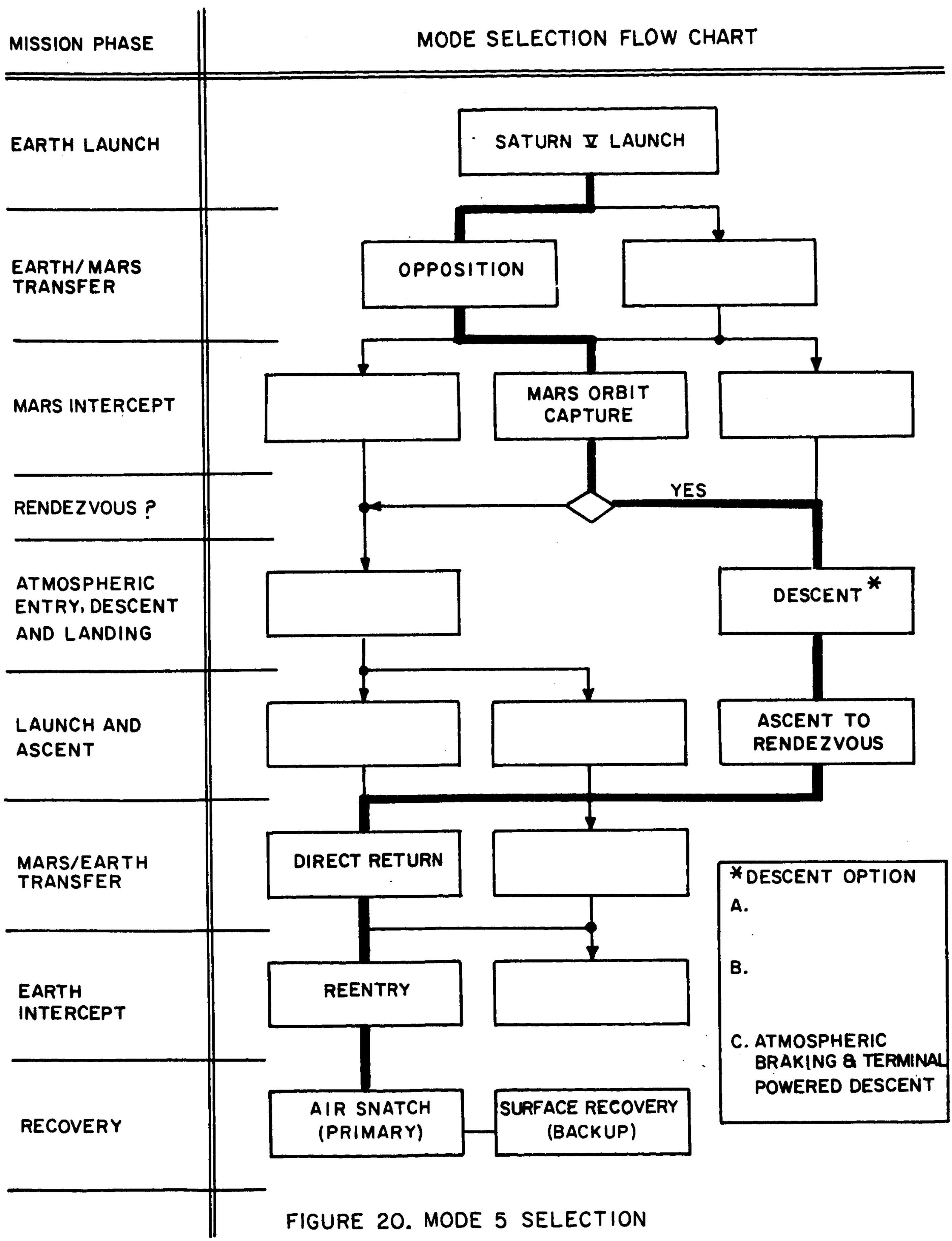

FIGURE 20. MODE 5 SELECTION

Mars landed weight	-	3,550 lb
Mars launch vehicle	-	3,020
Mars rendezvous capsule	-	440
Orbit trimmed weight	-	4,825
Mars/Earth spacecraft	-	630
Earth reentry container	-	55

The Saturn V injected spacecraft weight is 34,250 lb. The payload weight following orbit capture is 15,625 lb. Propulsion for both midcourse and orbit capture maneuvers uses cryogenic OF_2-B_2H_2 propellant, ISP = 400 sec. dV is 264 km/sec including 150 m/sec for midcourse corrections. The subsequent mission phases through rendezvous and docking are identical to Mode 3. However, because a higher energy direct Mars/Earth transfer follows, the rendezvous bus weight including escape propulsion is increased to 7,075 lb. Following docking the trimmed vehicle weight is 4,825 lb.

The return transfer is initiated just 2 days after Mars arrival. The injected spacecraft weighs 630 lb. Escape propulsion uses liquid storable N_2O_4-N_2H_4 propellant, ISP = 310 sec. Escape dV is 4.64 km/sec.

In order to minimize the Earth reentry speed of the direct return transfer the Mars departure date (and hence arrival date for stay time of 2 days) has to be as early as possible. The earliest Mars arrival date which can be used is February 21, 1976; outbound flight time is about 250 days. (This was iteratively determined, using Figure 1, until the maximum orbit weight was equal to the required orbit weight

15,625 lb.) Hence, the Mars departure date is February 23, 1976; return flight time is about 260 days. Total trip time is 512 days. The Earth reentry speed is almost 65,000 ft/sec.

Two midcourse maneuvers can be made during the return transfer, sharing a total midcourse capability of 200 m/sec. Propellant is N_2H_4, ISP = 235 sec. While a reentry phase similar to that used in all the other modes is postulated, the added 25,000 ft/sec entry speed will almost certainly have a profound effect on the Earth reentry capsule weight. No weight allowance was made, however, since no excess weight was available. A later Mars arrival date would have to be used to accommodate any increase in the Earth reentry capsule weight (presently 55 lb). This would delay the departure date (since a 2 day stay time is probably minimum) which would only further increase the reentry speed at Earth. Hence, both payload and design feasibility are doubtful. Mode 5 was presented to illustrate, by comparison with Mode 4, the necessity of Venus swingbys with short stay time.

5. SUMMARY AND CONCLUSIONS

A summary of the interplanetary transfers used for a weight analysis of mission Modes 1-5 is given in Table 2. The weight breakdown results of each mode is summarized in Table 3. The first four modes are well within the capability of a Saturn V launch vehicle based on the performance assumptions given in Table 1. Mode 5 is probably not feasible because of the high Earth reentry speed required. Furthermore,

Table 2

MISSION MODE TRAJECTORY SUMMARY

Parameter	Modes 1-3	Mode 4	Mode 5
Earth launch date	9/9/75	9/14/75	6/6/75
Outbound flight time, days	364	200	250
Mars arrival date	9/7/76	4/1/76	2/21/76
Approach VHP*, km/sec	2.72	4.05	3.52
Stay time, days	306	12	2
Departure VHP*, km/sec	2.98	5.81	6.38
Mars departure date	7/10/77	4/13/76	2/23/76
Return flight time	305	337	260
Venus miss altitude, km	-	1134	-
Earth arrival date	5/11/78	3/16/77	11/9/76
Inertial reentry speed, ft/sec	37,650	39,600	64,800
Total trip time, days	975	549	512

*Hyperbolic excess speed

Table 3

MISSION MODE WEIGHT* BREAKDOWN SUMMARY

Item	Mode 1	Mode 2	Mode 3	Mode 4	Mode 5
Saturn V capability	75,000	75,000	75,000	69,000	34,250
Total spacecraft	50,600	28,850	21,200	36,150	34,250
Mars orbit payload	26,900	---	11,250	14,450	15,625
Mars rendezvous bus	---	---	2,700	5,900	7,075
Mars entry capsule	22,300	26,750	7,100	7,100	7,100
Landed payload	8,920	8,920	3,550	3,550	3,550
Launch vehicle	7,960	7,960	3,020	3,020	3,020
Rendezvous capsule	---	---	440	440	440
Orbit trimmed weight	1,380	1,380	1,500	3,900	4,825
Return spacecraft	580	580	630	660	630
Earth reentry capsule	50	50	55	55	55

*All weights given in Earth lb. Weights do not include propulsion hardware weight of prior maneuvers.

any increase in the total spacecraft weight requirement would result in even higher Earth reentry speeds.

As a result of the study analysis a number of reservations regarding mission feasibility were apparent. Included are the following items:

- Payload descent ratios for Mars landed weights in excess of 5,000 lb have not been verified by optimized analysis (Modes 1 and 2),
- Adequate navigation, guidance and control capability for direct entry, heavy descent capsules has not been demonstrated (Mode 2),
- System and design requirements of unmanned rendezvous and docking in Mars orbit are virtually undefined (Modes 3 and 4),
- Structure factors for all propulsion maneuvers have not been verified by analysis of specific hardware systems (all modes),
- Guidance and control of a Venus swingby maneuver has been assumed feasible (Mode 4),
- Earth reentry of a 1-2 lb Mars sample with a 50 lb entry capsule for entry speeds up to 40,000 ft/sec has received only preliminary design consideration (all modes).

However, in view of the large payload contingencies noted in the first four mission mode descriptions (50 - 250 percent) there is a fair degree of certainty that their weight requirements will remain within the capability of a single Saturn V with continued analysis of these problems.

A number of specific subjects are recommended for continued study to further determine the value and feasibility of AMSR missions. These subjects include:

- Determination of the scientific objectives applicable to AMSR missions,
- Evaluation of the sample size, and the collection and storage requirements,
- Analysis of Mars landing site availability and fluctuation with launch opportunity and Mars intercept option,
- Investigation of changing weight requirements and new mission modes (e.g., outbound Venus swingbys) with later launch opportunities,
- Analysis of rendezvous and docking schemes and associated systems design requirements,
- Definition of optimum descent profiles for heavy (8-10,000 lb) Mars landers,
- Comparison of assumed structure factors with available propulsion system hardware and designs,
- Analysis of midcourse requirements with different transfer trajectories (including Venus swingby) and Mars intercept options,
- Determination of payload penalties of launch windows and plane change requirements.

The study results support a continued interest in AMSR missions which should be encouraged at least until these recommendations are satisfied.

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Appendix A

SUPPORT ANALYSIS

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SUPPORT ANALYSIS

A.1 SATURN V PERFORMANCE

The Saturn V injected spacecraft weight is shown in Figure A1 as a function of hyperbolic excess speed, V_{∞}. The data used to generate Figure A1 assumes a 90° azimuth Cape Kennedy launch to a 100 N.M. parking orbit (OSSA 1966). Performance is estimated for post-1970 launch vehicle growth. An effective shroud and adapter weight has been accounted for in determining the injected spacecraft weight.

A.2 MARS ORBIT SELECTIONS

The planetocentric orbits selected at Mars for the purpose of the study are summarized as follows:

Initial capture orbit	2.0 x 2.0 Mars radii
Entry ellipse	2.0 x 0.9
Parking orbit (direct launch)	1.1 x 1.1
Pre-rendezvous orbit (capsule)	1.1 x 1.3
Rendezvous transfer orbit (bus)	2.0 x 1.3
Rendezvous orbit	1.3 x 1.3

An initial circular capture orbit was selected to permit rapid observation and site selection as well as to allow

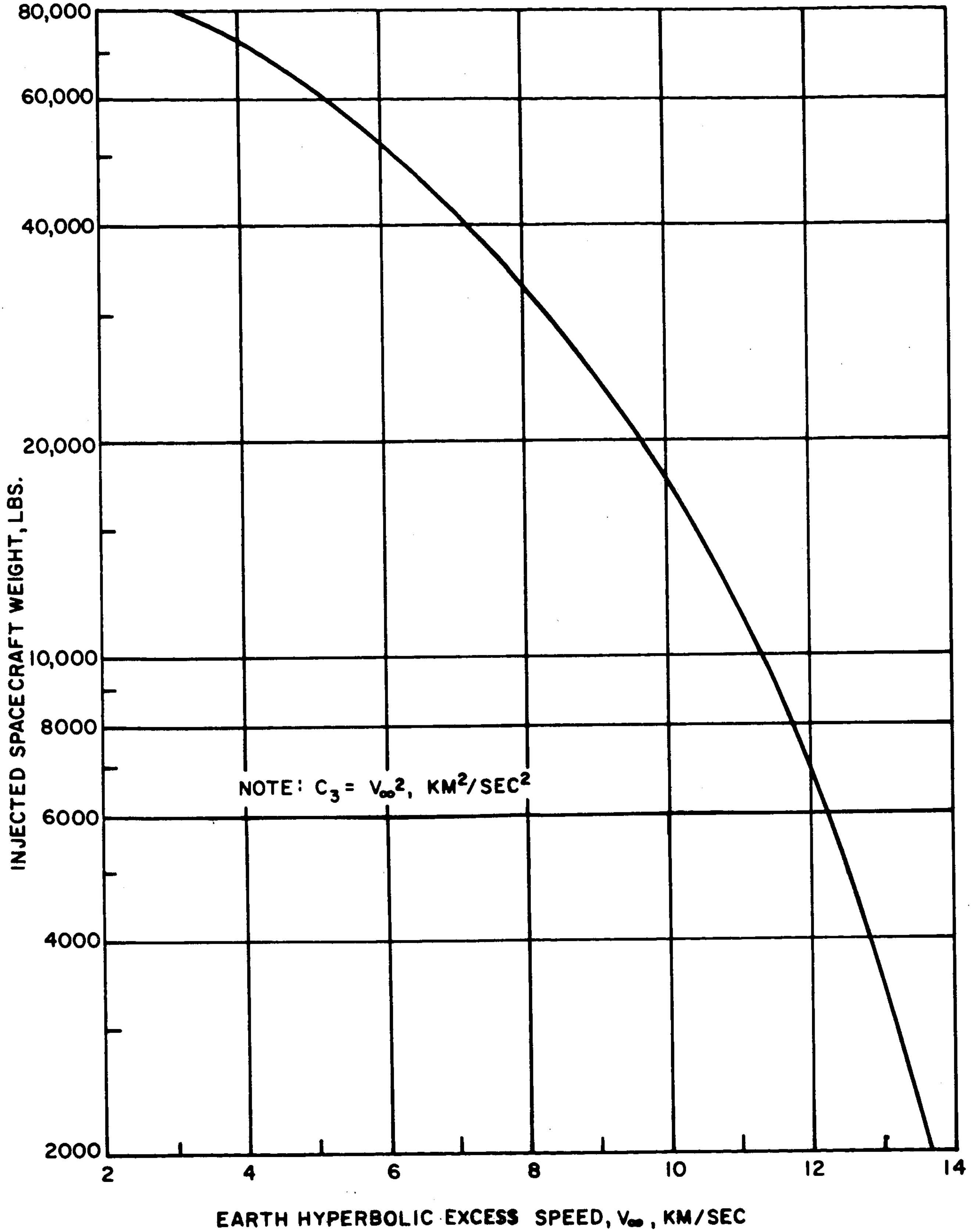

FIGURE AI. INJECTED SPACECRAFT WEIGHT CAPABILITY FOR THE SATURN V LAUNCH VEHICLE the deorbit trajectory of the entry capsule to be independent of the position in orbit. Circular orbits also provide an important simplification to Mars orbit rendezvous. If both the capture orbit of the spacecraft upon arrival at Mars and the parking orbit of the rendezvous capsule after launch from Mars are circular, then rendezvous is independent of apsline.

Other factors which affect the selection of the altitude of the circular capture orbit are (1) low altitudes provide the best possible photographic resolution for landing site selection, (2) higher altitudes guarantee that dispersions in the aiming point at Mars (due to orbit determination uncertainty and midcourse correction errors) do not result in penetration of the sterilization altitude assumed to be 1000 km (Tarver 1967), and (3) if rendezvous is to be used, larger altitude differences between the final rendezvous orbit (1.3 Mars radii) and the capture orbit provide more frequent opportunities for rendezvous (see Section A.6).

The first concern of capture orbit radius selection is to retro at the optimum radius so that the capture dV requirement is minimized. The equation for dV_{retro} to a circular orbit is

$$dV = \left(VHP^2 + \frac{2K}{r}\right)^{1/2} - \left(\frac{K}{r}\right)^{1/2} .$$

where K is the gravitational parameter of Mars and VHP is the hyperbolic approach velocity at Mars. Differentiating with respect to r and setting the results to zero yields

$$0 = \frac{K}{r^2}\left(\frac{1}{2\sqrt{\frac{K}{r}}} - \frac{1}{\sqrt{VHP^2 + \frac{2K}{r}}}\right) .$$

Solving then for the circular orbit radius, r which requires the minimum capture dV

$$r = \frac{2K}{VHP^2} .$$

The values of VHP, for the Earth/Mars transfers considered, vary from about 2.75 to 4 km/sec. The corresponding circular orbit radii for minimum capture dV are 3.4 to 1.6 Mars radii, respectively.

The circular capture orbit radius selected was 2.0 Mars radii, approximately mid-way between the range for minimum dV capture. This was determined primarily by aiming point dispersions. For a 2 Mars radii capture maneuver (assuming the maneuver is made at periapse of the approach hyperbola) the maximum acceptable dispersion in the aiming point is approximately 2400 km due to the 1000 km sterilization altitude. To insure that there be only one chance in 10,000 (4σ) of violating the sterilization constraint the tolerable 1σ dispersion in aiming point would be 600 km. For a lesser requirement of one chance in 100 (3σ) of contamination a 1σ aiming point error of 800 km would be acceptable.

These values were compared with statistical error analysis results for Mars trajectories which are presented in the recent Mars Probe Study (AVCO 1966a). The 1σ error in aiming point for a Type I (opposition) trajectory for the 1971 launch period reportedly varied from 575 to 760 km. The 1σ error for a Type II (conjunction) trajectory for the 1975 launch period varied from 1300 to 1500 km. The proposed 1σ value of 600 km is in the range of the Type I trajectory data. The 1σ values of the Type II trajectory data, however, are more than double the required value, and seem unduly high. Further study will be necessary to resolve whether 1500 km is a realistic estimate of 1σ Mars aiming point errors of conjunction type transfers of mid-1970 Mars missions. If it is, the selected capture orbit radius of 2 Mars radii would have to be raised to guarantee the required probability of only 1 chance in 10,000 of contamination.

Added to the aiming point errors is another possibility of contamination due to dispersions in the retro velocity of the capture maneuver. However, for a reasonable estimate in velocity dispersions it can be shown that the change in orbit periapse and apoapse radii is small over the range of aiming point dispersions. This is illustrated in Figure A2. Plotted as a function of periapse radius of the approach hyperbola is the dispersion in the counter apse point of the capture orbit for an error in capture velocity of 10 m/sec.* The dispersion varies from about 100 to 120 km, more than an order of magnitude less than the aiming point error dispersion limit (2400 km).

*10 m/sec corresponds to a 0.5 percent error in capture dV of a conjunction type Earth/Mars transfer.

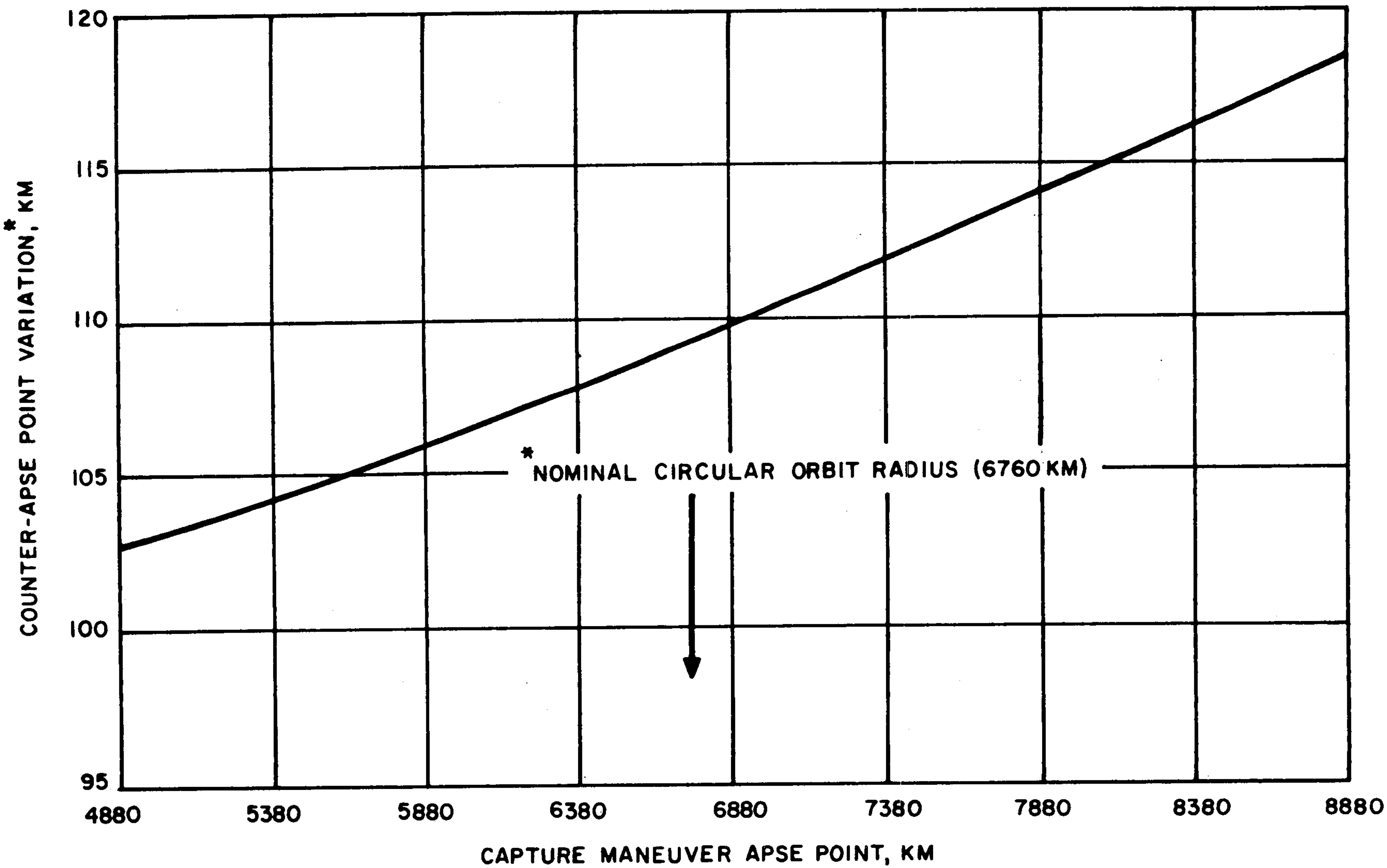

FIGURE A2. CAPTURE ORBIT VARIATION

The entry ellipse, 2.0 x 0.9 Mars radii, was selected to approximate the entry conditions used for entry analysis which were as follows:

Entry altitude	= 50 km
Entry angle (γ)	$\doteq$ -15°
Entry velocity (V_E)	= 3.8 km/sec

The parking orbits used for launch vehicle injection conditions were selected as 1.1 Mars radii circular for direct launch and 1.1 x 1.3 for launch to rendezvous (so that only one additional burn would be necessary to place the rendezvous capsule in the 1.3 circular rendezvous orbit). The effect on launch vehicle performance of changing the orbit low points from 1.1 to 1.05 was studied. Preliminary results indicated little change in performance so that initially selected higher value, 1.1, was retained.

The rendezvous transfer orbit, 2.0 x 1.3 Mars radii, which is used by the rendezvous but to initiate rendezvous from its 2 radii capture orbit is a minimum energy Hohmann transfer to the rendezvous capsule orbit of 1.3 Mars radii circular.

A.3 INTERPLANETARY TRAJECTORY ANALYSIS

A single data source has been used for Earth/Mars single-plane transfer trajectories (Planetary Flight Handbook 1963). For direct entry (no capture orbit) at Mars the hyperbolic Earth excess speed, V_∞, was minimized for transfer times of less than one year* in order to maximize the available Mars entry weight. This was done for Mars arrival dates between Earth-Mars opposition and conjunction. Parametric trajectory data and transfer weights are tabulated in Table A-1 for the 1975 and 1977 launch opportunities. A graph of the 1975 data is presented in Figure A3. A graph of the 1977 data is plotted in Figure A4.

For orbit capture at Mars, the payload weight in a 2 Mars radii circular orbit was maximized for transfer times of less than one year.* This was done by varying the launch date for a given arrival date until V_∞ (at Earth) and VHP (at Mars), combined with Saturn V performance (Figure A1) and capture engine performance (ISP = 400 sec, f = 0.111) respectively, yielded maximum orbit weight. Midcourse corrections during the Earth/Mars transfer (150 m/sec) were included with the computed impulsive capture velocity requirement. Parametric trajectory data and orbit weights are tabulated in Table A-2 for the 1975 and 1977 launch opportunities. A graph of the 1975 data is presented in Figure A5. A graph of the 1977 data is plotted in Figure A6.

Data for the return Mars/Earth transfer trajectories were calculated on the IITRI IBM 7094 digital computer using the JPL Space Research Conic Program (Joseph and Richard 1966) In order to minimize the weight requirements for Mars departure

*The one year flight time constraint was relaxed as arrival dates approached Earth/Mars conjunction in order to maintain payload weight continuity.

TABLE A-1

MARS/EARTH TRANSFER DATA FOR DIRECT MARS ENTRY

Julian Arrival Date	Calendar Arrival Date	Flight Time Days	Earth V_{∞} km/sec	Saturn V Injected wt., lb.	Mars VHP
244 2760	12-13-75	160	9.95	17,700	9.21
2790	1-12-76	130	7.15	40,200	9.92
2820	2-11-76	150	5.30	59,600	7.18
2850	3-12-76	180	4.53	68,500	5.04
2880	4-11-76	210	4.32	70,500	3.66
2900	5-1-76	230	4.50	69,000	3.16
2950	6-20-76	310	4.56	68,000	2.53
3000	8-9-76	340	3.84	76,000	2.53
3050	9-28-76	380	3.66	78,000	2.95
3100	11-17-76	420	3.75	77,000	3.75
244 3530	1-21-78	220	11.26	10,200	6.47
3560	2-20-78	130	7.75	35,000	12.28
3590	3-22-78	150	5.51	57,500	8.94
3620	4-21-78	190	4.53	68,500	6.14
3650	5-21-78	220	4.20	72,000	4.47
3700	7-10-78	290	4.02	74,000	3.19
3750	8-29-78	330	3.31	81,000	2.44
3800	10-18-78	370	3.34	80,500	3.01
3850	12-7-78	400	3.55	79,000	4.17
3900	1-26-79	430	3.72	77,000	5.69

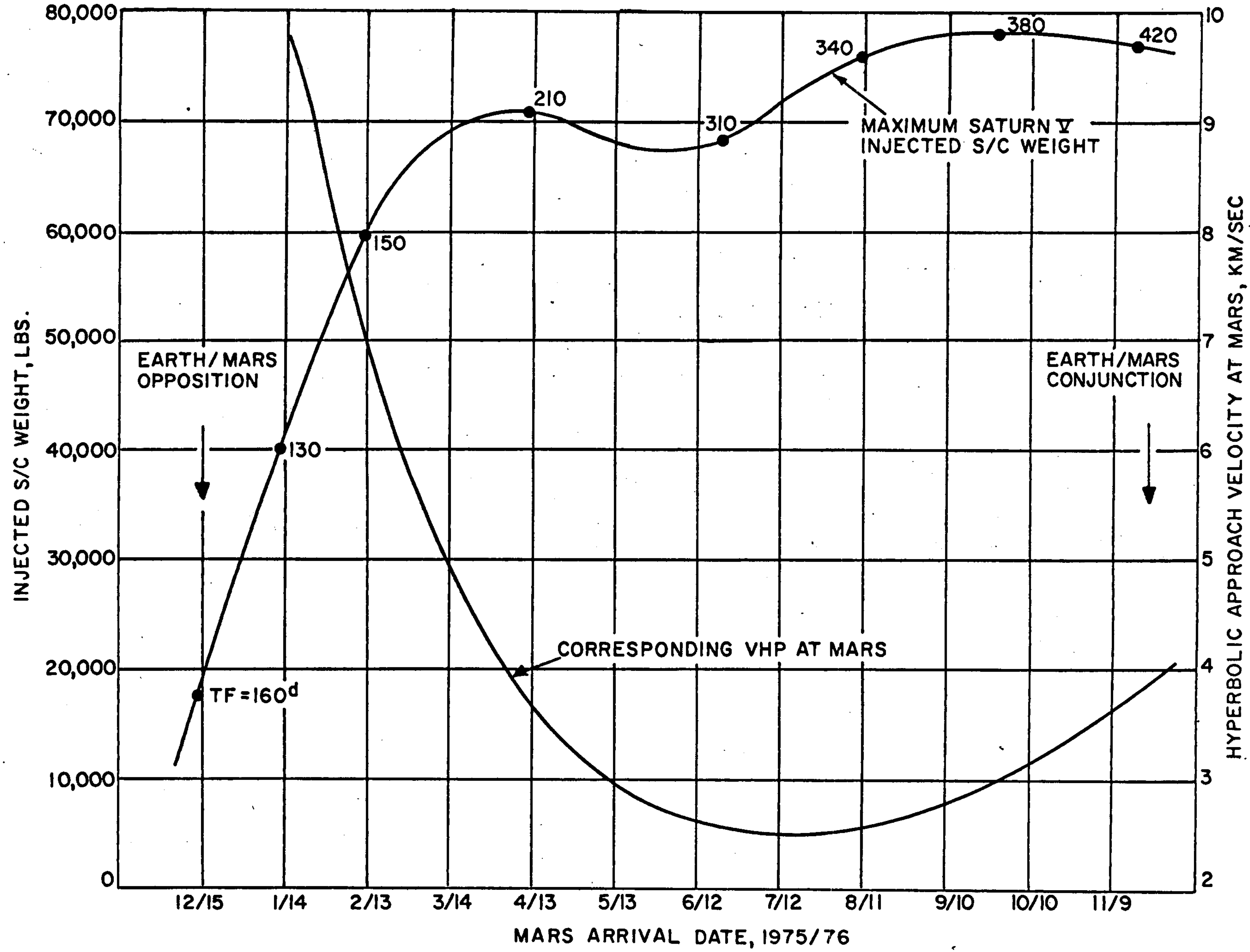

FIGURE A3. MAXIMUM OUTBOUND TRANSFER PAYLOAD

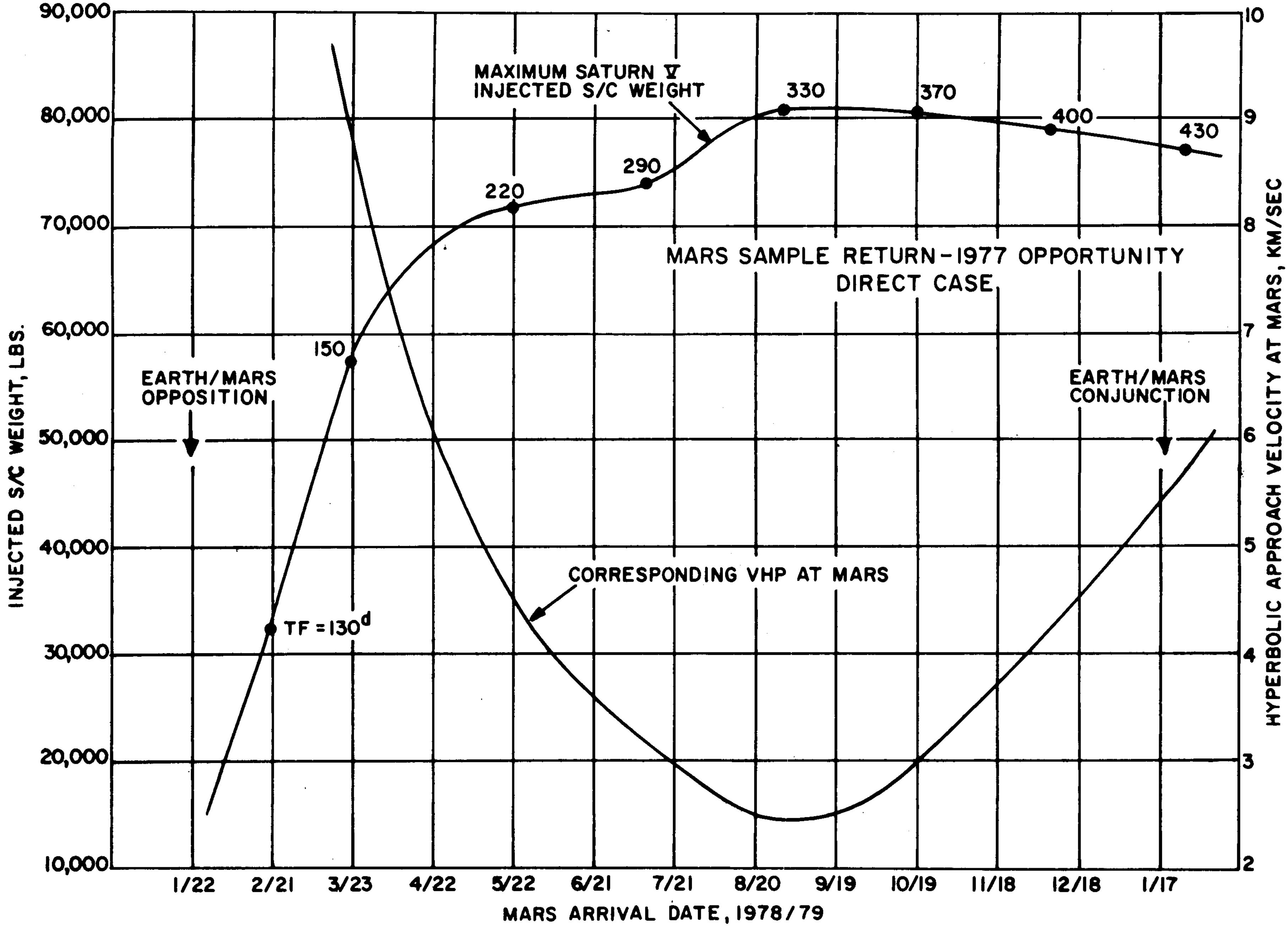

FIGURE A4. MAXIMUM OUTBOUND TRANSFER PAYLOAD

TABLE A-2

EARTH/MARS TRANSFER AND MAXIMUM ORBIT* WT. DATA

Julian Arrival Date	Calendar Arrival Date	Flight Time, Days	Earth V_∞ km/sec	Saturn V Injected Wt. Lbs.	Mars VHP km/sec	Total dV (capture & m/c) km/sec	Mass Fraction ISP f 400 0.111	Maximum Wt. in Orbit, 1
244 2760	12-13-75	240	10.25	16,000	4.23	3.18	0.381	6,096
2790	1-12-76	240	9.24	22,800	3.99	2.99	0.406	9,257
2820	2-11-76	250	8.25	30,700	3.55	2.67	0.450	13,815
2850	3-12-76	190	4.62	66,600	4.92	3.71	0.318	21,179
2860	3-22-76	190	4.41	68,800	4.50	3.37	0.359	24,699
2880	4-11-76	210	4.32	69,700	3.67	2.75	0.436	30,389
2900	5-1-76	220	4.53	67,600	3.04	2.33	0.500	33,800
2950	6-20-76	310	4.56	67,300	2.53	2.02	0.550	37,015
3000	8-9-76	340	3.84	74,000	2.53	2.02	0.550	40,700
3050	9-28-76	380	3.67	75,400	2.95	2.28	0.509	38,379
3100	11-17-76	410	3.78	74,500	3.73	2.80	0.430	32,035
244 3530	1-21-78	240	11.56	8,800	5.16	3.90	0.299	2,631
3560	2-20-78	250	10.31	15,600	4.77	3.59	0.333	5,195
3590	3-22-78	260	8.94	25,100	4.38	3.28	0.370	9,287
3620	4-21-78	200	4.80	64,800	5.99	4.62	0.230	14,904
3650	5-21-78	220	4.20	70,700	4.47	3.34	0.361	25,523
3700	7-10-78	300	4.08	71,900	2.86	2.20	0.520	37,388
3750	8-29-78	330	3.31	78,300	2.44	1.96	0.560	43,848
3800	10-18-78	370	3.34	78,100	3.01	2.31	0.501	39,128
3850	12-7-78	400	3.55	76,500	4.17	3.12	0.390	29,835
3900	1-26-79	430	3.73	74,800	5.69	4.35	0.255	19,074

* Mars orbit - circular, 2 Mars radii

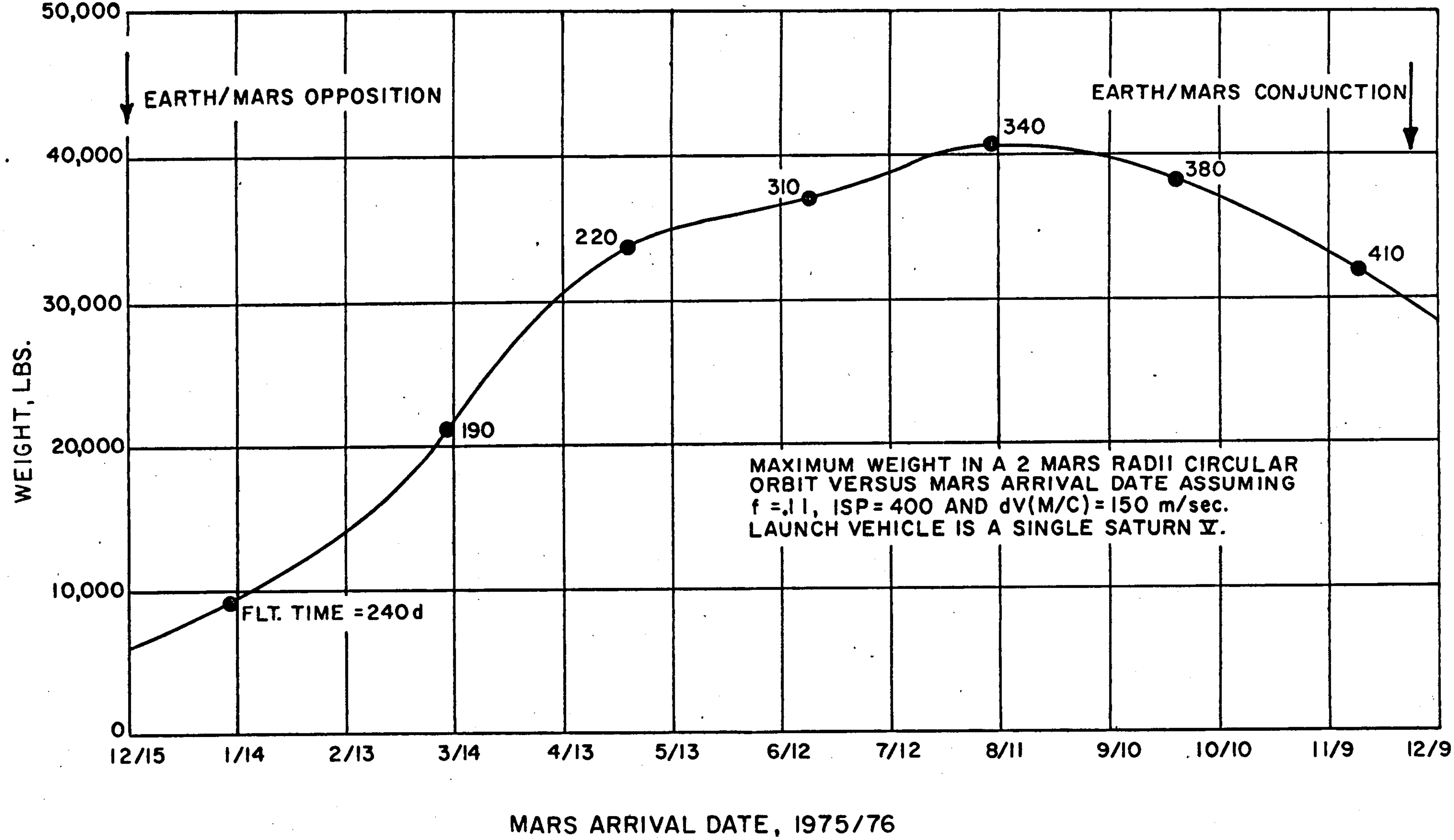

FIGURE A5. MAXIMUM MARS ORBIT PAYLOAD CAPABILITY of the VM-8 and VM-2 atmospheres shown in Table A-6. Emphasis has been placed on the VM-8 worst case approximation. The atmosphere was assumed to be strictly isothermal with a constant scale height. A few calculations were made assuming a two layer atmosphere with a discontinuity at the tropopause, but with constant scale height within each layer.

The range of ballistic coefficients used was from 0.5 to 3 slugs/ft^2. The lower value is compatible with entry vehicles less than 5000 lb in weight. In particular it overlaps values which had been used in Voyager atmospheric probe studies (Pragluski 1966, AVCO 1966a and b).

A set of lift to drag ratios of 0.0 - 1.0 were considered. A value of 0.25 is typical of Gemini reentry; a value of 0.5 is typical of lifting conic shapes and a value of 1.0 (or greater) is characteristic of lifting bodies.

Entry was taken to start at an altitude of 50 km. As the deceleration due to atmospheric friction was found to just about balance the gravity gains above this value, 50 km was used to suppress output and to conserve computing time.

Solutions to the entry equations were obtained in two ways:

(a) Using numerical integration routines contained in a set of 4th order Runge-Kutta integration subroutines,

(b) through an exact solution due to Loh (1963) for constant angle of entry.

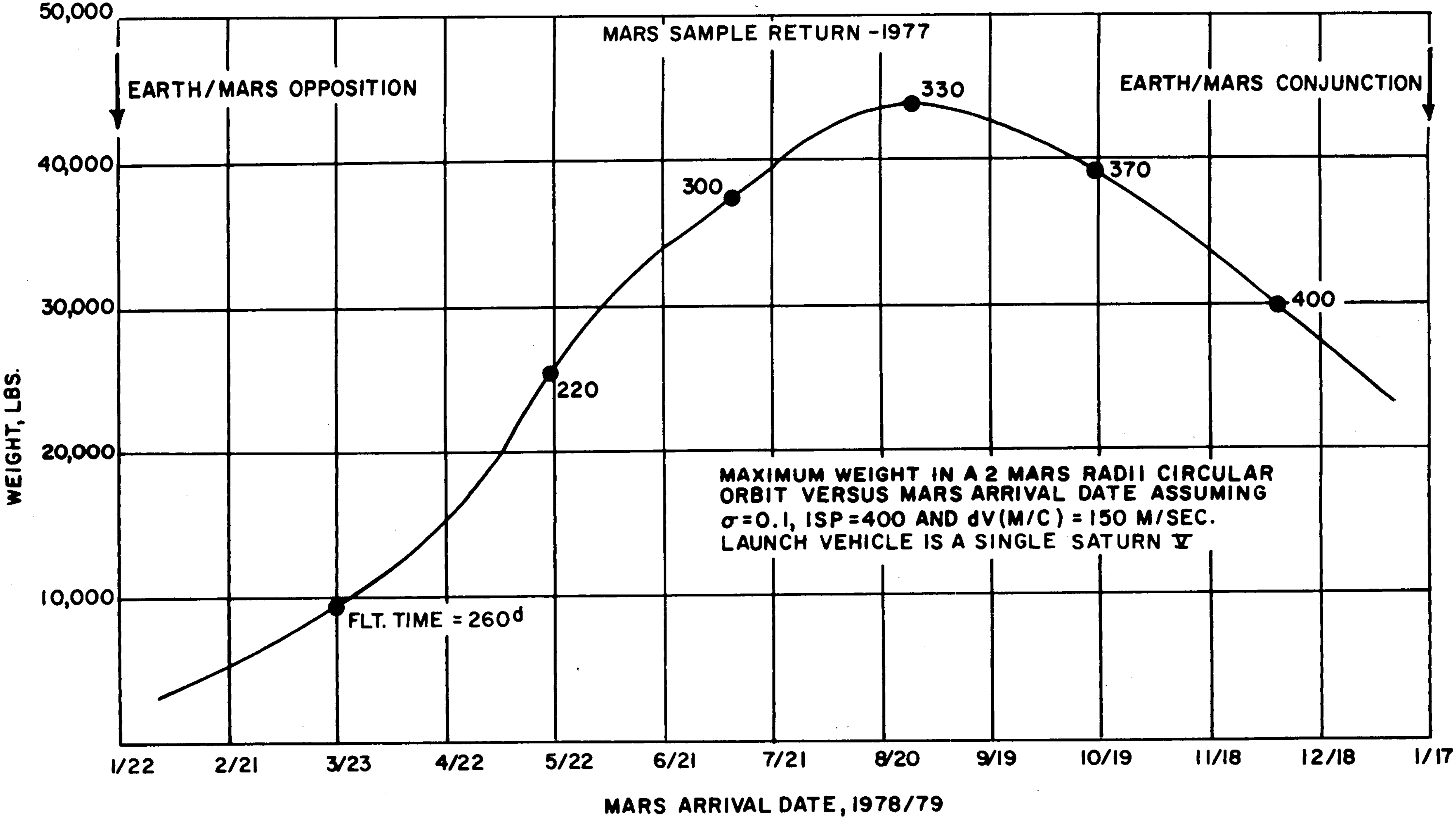

FIGURE A6. MAXIMUM MARS ORBIT PAYLOAD CAPABILITY only minimum energy return transfers were computed. An ISP = 310 sec and a structure factor f = 0.111 were assumed for weight calculations. It turns out that the in orbit weight requirement is approximately the same for the 680 lb spacecraft returning from the 1.3 radii rendezvous orbit as for the 580 lb spacecraft returning from the 1.1 radii parking orbit.

Parametric trajectory data, required weights and Earth reentry speeds are tabulated in Table A-3 for short Mars stay times compatible with the 1975 and 1977 launch opportunities (at Earth). A graph of the required orbit weight and Earth reentry speed for short stay times compatible with the 1975 opportunity is plotted in Figure A7. This data for the 1977 opportunity are also presented in Figure A8.

Long stay times at Mars make available minimum energy return transfers. Parametric trajectory data, required orbit weights, and Earth reentry speeds for these return transfers are tabulated in Table A-4. The 1977 Mars departure dates are compatible with the 1975 opportunity Mars arrival dates and stay times of up to one year. A graph of required orbit weight and Earth reentry speed versus Mars departure date for long stay time is presented in Figure A9.

Venus swingby return transfers were also considered for 1975 short stay time missions. Mars departure dates during the first half of 1976 were investigated. Minimum Earth reentry speeds were found when Type I Class 1 Mars/Venus trajectories and Type II Class 2 Venus/Earth trajectories are used. A Mars

TABLE A-3

DIRECT RETURN DATA FOR SHORT STAY TIME

Mars Calendar Departure Date	Flight Time Days	Mars VHL km/sec	Escape dV from 1.3 Circ. Orbit km/sec	Mass Fraction ISP f 310 0.111	Required Orbit wt. to Return 630 lbs.	Earth VHP km/sec	Entry Speed* at 400,000 ' ft/sec
12-19-75	287	5.23	3.73	0.215	2,930	12.80	55,400
1-28-76	271	5.89	4.27	0.162	3,889	15.05	61,100
3-8-76	254	6.67	4.91	0.110	5,727	17.20	67,000
4-17-76	236	7.57	5.67	0.062	10,161	19.21	72,700
5-7-76	228	8.08	6.10	0.038	16,579	20.18	75,500
5-27-76	219	8.62	6.56	0.018	35,000	21.11	78,200
1-21-78	284	4.93	3.50	0.242	2,603	13.55	57,300
3-22-78	257	5.96	4.31	0.158	3,987	16.22	64,300
5-21-78	231	7.25	5.40	0.077	8,182	18.54	70,800
6-20-78	218	7.99	6.02	0.042	15,000	19.53	73,500
7-20-78	206	8.82	6.74	0.013	48,462	20.41	76,100

* Inertial entry speed.

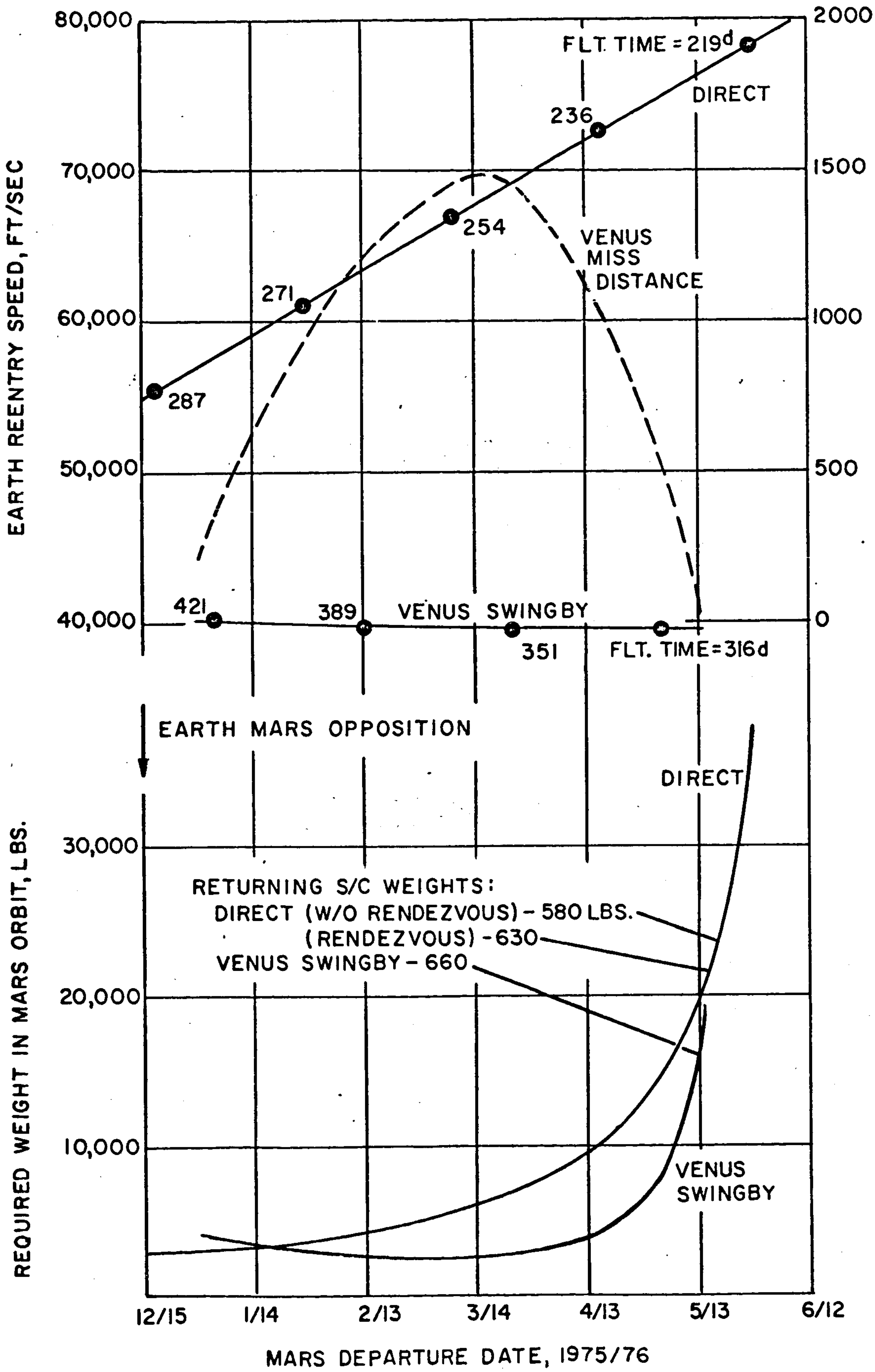

FIGURE A7. REQUIRED ORBIT PAYLOAD FOR RETURN TRANSFER, SHORT STAY TIME

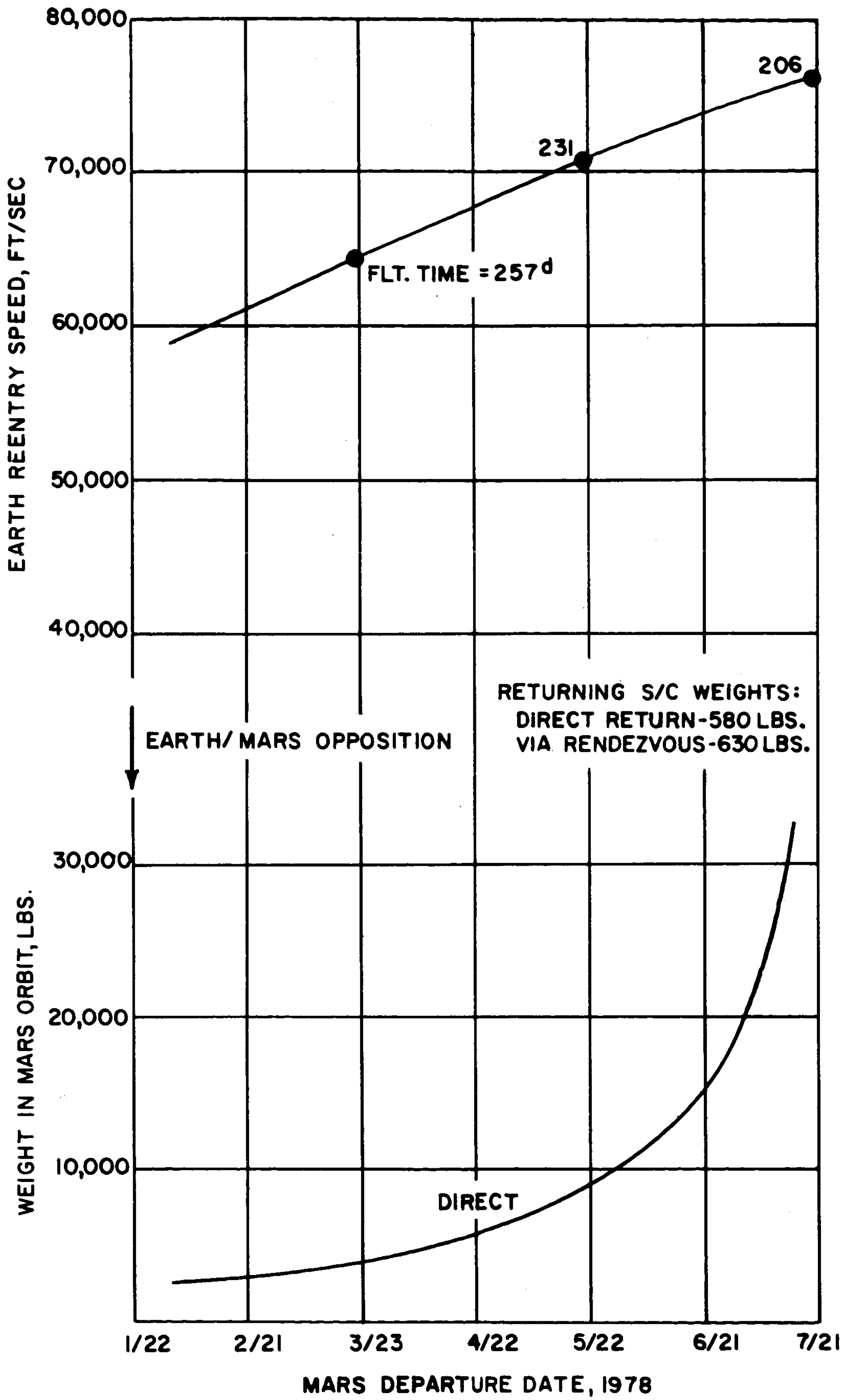

FIGURE A8. REQUIRED ORBIT PAYLOAD FOR RETURN TRANSFER, SHORT STAY TIME.

TABLE A-4

DIRECT RETURN DATA FOR LONG STAY TIME

Mars Calendar Departure Date	Flight Time Days	Mars VHL km/sec	Escape dV from 1.3 Circ. Orbit km/sec	Mass Fraction ISP 310 f 0.111	Required Orbit Wt. to Return 630 Lbs.	Earth VHP km/sec	Entry Speed* at 400,000' ft/sec
5-1-77	353	4.88	3.46	0.247	2551	3.09	37,850
5-21-77	341	4.22	2.98	0.303	2079	3.02	37,825
6-10-77	328	3.62	2.60	0.362	1740	2.95	37,800
6-30-77	313	3.14	2.31	0.407	1548	2.89	37,700
7-20-77	296	2.88	2.17	0.432	1458	2.85	37,600
8-9-77	281	2.96	2.20	0.425	1482	2.80	37,500
8-29-77	316	3.21	2.34	0.401	1571	4.93	39,800
9-18-77	321	3.42	2.48	0.380	1658	6.51	42,200
9-28-77	321	3.52	2.53	0.371	1698	7.21	43,300

* Inertial entry speed.

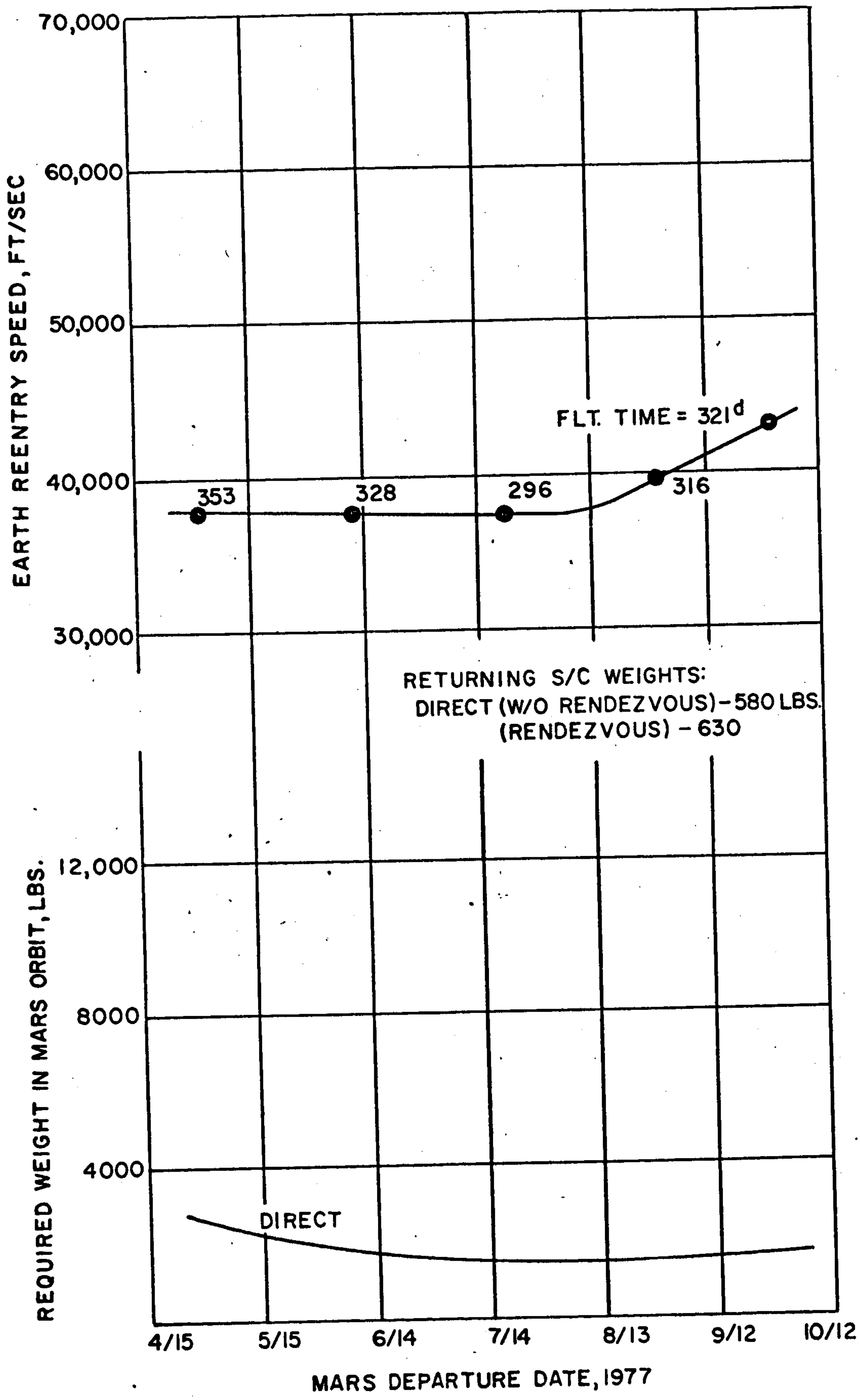

FIGURE A9. REQUIRED ORBIT PAYLOAD FOR RETURN TRANSFERS, LONG STAY TIME hyperbolic departure velocity was selected for these type transfers such that the miss distance at Venus was maximized. Parametric trajectory data, required orbit weights and Earth reentry speeds are tabulated in Table A-5. These results are presented in Figure A7.

A.4 ATMOSPHERIC ENTRY

The first cursory appraisal of atmospheric entry at Mars indicated the entry mode should combine atmospheric braking with a powered terminal descent. However, no information was available on the entry of heavy landers (> 5000 lb) using the recent low surface pressure of 5 mb. This section discusses a preliminary parametric analysis of the entry characteristics in an attempt to derive approximate overall descent weight ratios.

The equations of motion describing the ballistic entry trajectory are

$$\frac{dV}{dt} = -\frac{1}{2} \rho \frac{V^2}{B} - \frac{K}{r^2} \sin \gamma ,$$

$$\frac{d\gamma}{dt} = \frac{1}{2} \rho \frac{V^2}{B} (L/D) - \frac{\cos \gamma}{r} \left(\frac{K}{r} - V^2\right) ,$$

where

γ = inertial flight path angle,
V = velocity of entry vehicle,
r = planetocentric radius vector of vehicle,
K = gravitational constant,
ρ = atmospheric density,
B = ballistic coefficient (M/C_dA),
L/D = lift to drag ratio,
M = entry vehicle weight,

Table A-5

VENUS SWINGBY RETURN DATA FOR SHORT STAY TIME

Mars Launch Date	Flight Time, days	Mars VHL, km/sec	Escape dV from 1.3 Circ. Orbit, km/sec	Mass Fraction ISP f 310 0.111	Required Orbit Wt. to Return 660 lb	Venus Miss Distance, km	Earth VHP, km/sec	Entry Speed* at 400,000', ft/sec
1-4-76	421	5.79	4.18	0.170	3882	366	5.19	40,150
1-24-76	404	5.25	3.75	0.213	3099	867	5.09	40,000
2-13-76	389	4.89	3.48	0.242	2727	1212	4.99	39,850
3-4-76	373	4.79	3.40	0.253	2609	1393	4.92	39,700
3-24-76	351	5.05	3.60	0.229	2882	1466	4.91	39,700
4-13-76	337	5.81	4.20	0.168	3929	1134	4.85	39,600
5-3-76	316	7.15	5.31	0.085	7765	546	4.81	39,550
5-14-76	305	8.17	6.18	0.034	19412	22	4.80	39,550

*Inertial entry speed

C_d = aeroshell drag coefficient,
A = area of aeroshell.

The entry angles (γ) into the atmosphere considered were between 90° to 10°. In general, the smaller entry angle the greater the use that can be made of atmospheric braking but a limit is reached where the entry vehicle will skip out of the atmosphere. Values of γ_o between -10° and -20°, specified at an altitude of 50 km, have been given most consideration. It is important that this entry altitude be specified since angles quoted for either deorbit or "vacuum entry" conditions may enter the sensible atmosphere at an angle considerably less than that quoted. For example, a γ = -20 at an altitude of 250 km and a velocity of 5.8 km/sec becomes reduced to γ = 10° at an altitude of 50 km.

The entry velocities considered range from 3.6 to 7.1 km/sec at 50 km altitude. The lower velocity corresponds to descent from near-minimum orbit while the higher velocity corresponds to direct entry from an interplanetary trajectory with an approach velocity (VHP) of 5 km/sec. Entry velocities of 4.0 km/sec and 5.8 km/sec have been assumed as typical of descent from orbit and direct entry.

Atmospheric densities of $\rho = 1.32 \times 10^{-2}$ kg/m^3 and $\rho = 1.85 \times 10^{-2}$ kg/m^3 have been assumed corresponding to surface pressures of 5 mb and 7 mb, respectively. A corresponding tropospheric scale height (H) of 10 km has been used throughout the study. Thus the atmospheric parameters approximate those of the VM-8 and VM-2 atmospheres shown in Table A-6. Emphasis has been placed on the VM-8 worst case approximation. The atmosphere was assumed to be strictly isothermal with a constant scale height. A few calculations were made assuming a two layer atmosphere with a discontinuity at the tropopause, but with constant scale height within each layer.

The range of ballistic coefficients used was from 0.5 to 3 slugs/ft^2. The lower value is compatible with entry vehicles less than 5000 lb in weight. In particular it overlaps values which had been used in Voyager atmospheric probe studies (Pragluski 1966, AVCO 1966a and b).

A set of lift to drag ratios of 0.0 - 1.0 were considered. A value of 0.25 is typical of Gemini reentry; a value of 0.5 is typical of lifting conic shapes and a value of 1.0 (or greater) is characteristic of lifting bodies.

Entry was taken to start at an altitude of 50 km. As the deceleration due to atmospheric friction was found to just about balance the gravity gains above this value, 50 km was used to suppress output and to conserve computing time.

Solutions to the entry equations were obtained in two ways

(a) Using numerical integration routines contained in a set of 4th order Runge-Kutta integration subroutines,

(b) Through an exact solution due to Loh (1963) for constant angle of entry.

Table A-6

BASIC ATMOSPHERE PARAMETERS*

Atmosphere	Surface Pressure (mb)	Surface Density ($kg/m^3 \cdot 10^{-2}$)	Scale Height Troposphere (km)	Stratosphere Altitude (km)	Troposphere Altitude (km)
VM-1	7.0	0.955	23.2	14.2	19.3
VM-2	7.0	1.85	10.0	5.0	18.6
VM-3	10.0	1.365	23.3	14.2	19.3
VM-4	10.0	2.57	10.8	5.1	17.1
VM-8	5.0	0.68	23.2	14.2	19.3
VM-8	5.0	1.32	10.0	5.0	18.6

*Pragluski (1966).

The solution for constant angle of entry implies that some non-zero L/D will be required. However, comparison of results from (a) and (b) indicated that constant angle is a good approximation for L/D values of the order of 0.1 or less. As results were obtained considerably faster with (b) than (a) the short time scale of the study dictated the use of (b) wherever possible.

The residual weight fraction of the atmospheric braking portion of the descent was determined by describing the aeroshell weight as a percent (ASF) of the total entry weight. Assuming the aeroshell is staged just before terminal powered descent the residual fraction for atmospheric braking (RAB) is given as

$$RAB = 1 - ASF$$

Residual weight fractions of the powered portion of descent were computed on the basis of a characteristic dV equal to the residual capsule velocity (i.e., an impulsive maneuver) at ignition altitudes (h) of 3 and 5 km. These retro weight fractions (RWF) were calculated from the equation

$$RWF = 1 - (1 + f)\left(1 - e^{-\frac{dV}{gISP}}\right)$$

where the specific impulse (ISP) was specified as 310 sec and the propulsion hardware factor was set at 0.15. The estimated error in dV for the impulsive approximation is less than 5 percent for the cases analyzed.

The overall descent fraction (ODF) of landed weight* to entry weight is given by the expression

$$ODF = (RAB)\ (RWF)$$

The effect of aerodynamic heating upon entry was estimated using the equation

$$Q = C_1\ \rho^{0.5}\ V^{3.04}\ \text{cals/cm}^2\text{-sec}\ .$$

This empirical equation has been derived from a previous entry heating study (Gilligan 1967) and was scaled by the factor C_1 to account for a pure CO_2 atmosphere. The factor also contains the dependence on the radius of curvature of the aeroshell.

The initial velocity for direct entry into the Mars atmosphere is shown in Figure A10 as a function of the approach velocity to Mars (VHP). The average entry velocity for out-of-orbit entry is about 4.0 km/sec.

Figure A11 shows time profiles of deceleration and velocity for direct 90° entry. Even for a small ballistic coefficient (B = 0.6), the terminal velocity is on the order of 3 km/sec. It has been concluded that the Mars atmosphere is too tenuous to provide significant drag for steep entry angles. This is borne out by the relatively small g loadings indicated in Figure A11.

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*Landed weight does not include either the aeroshell weight or the terminal descent propulsion hardware weight.

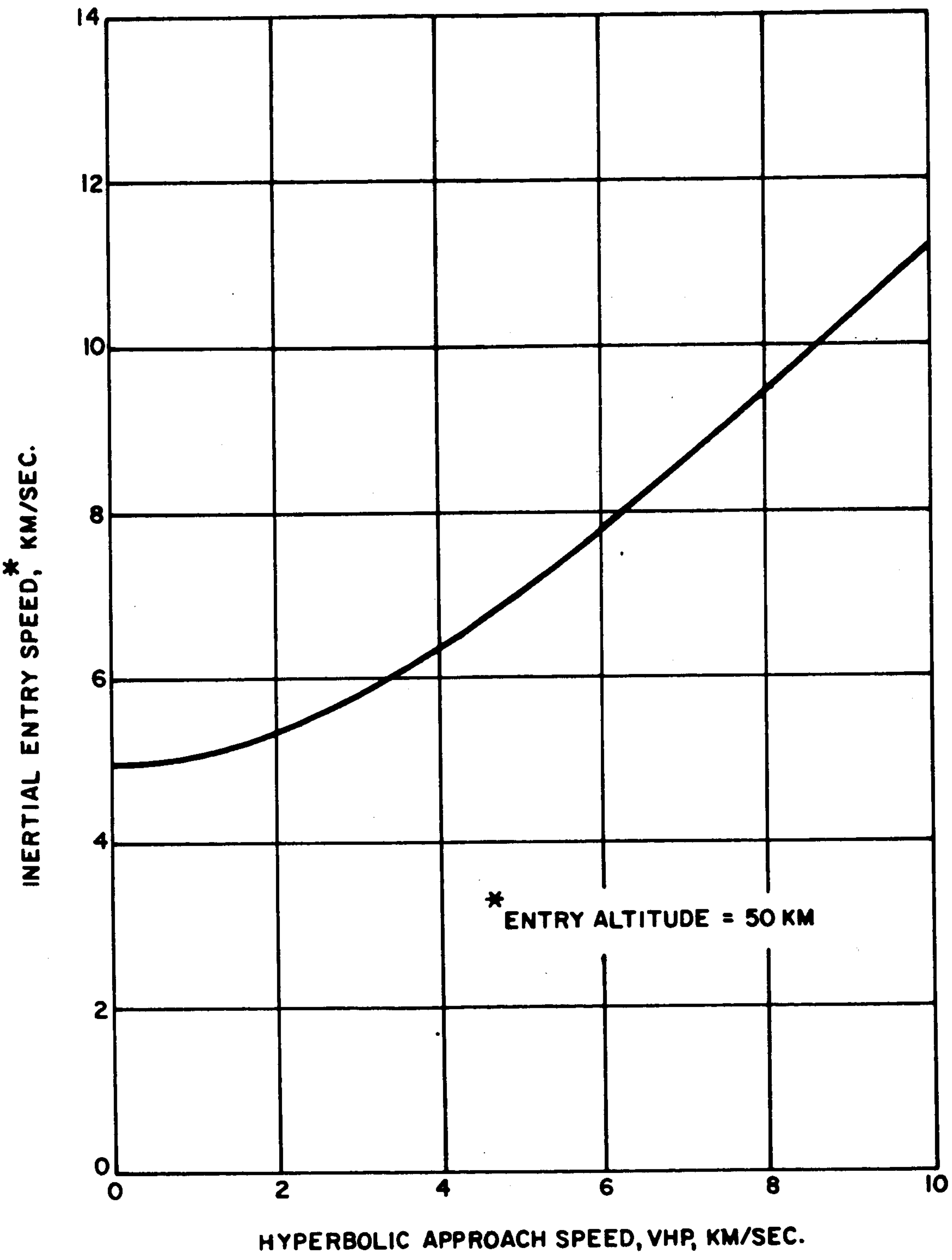


FIGURE A10. MARS ATMOSPHERIC ENTRY SPEED FOR DIRECT DESCENT.

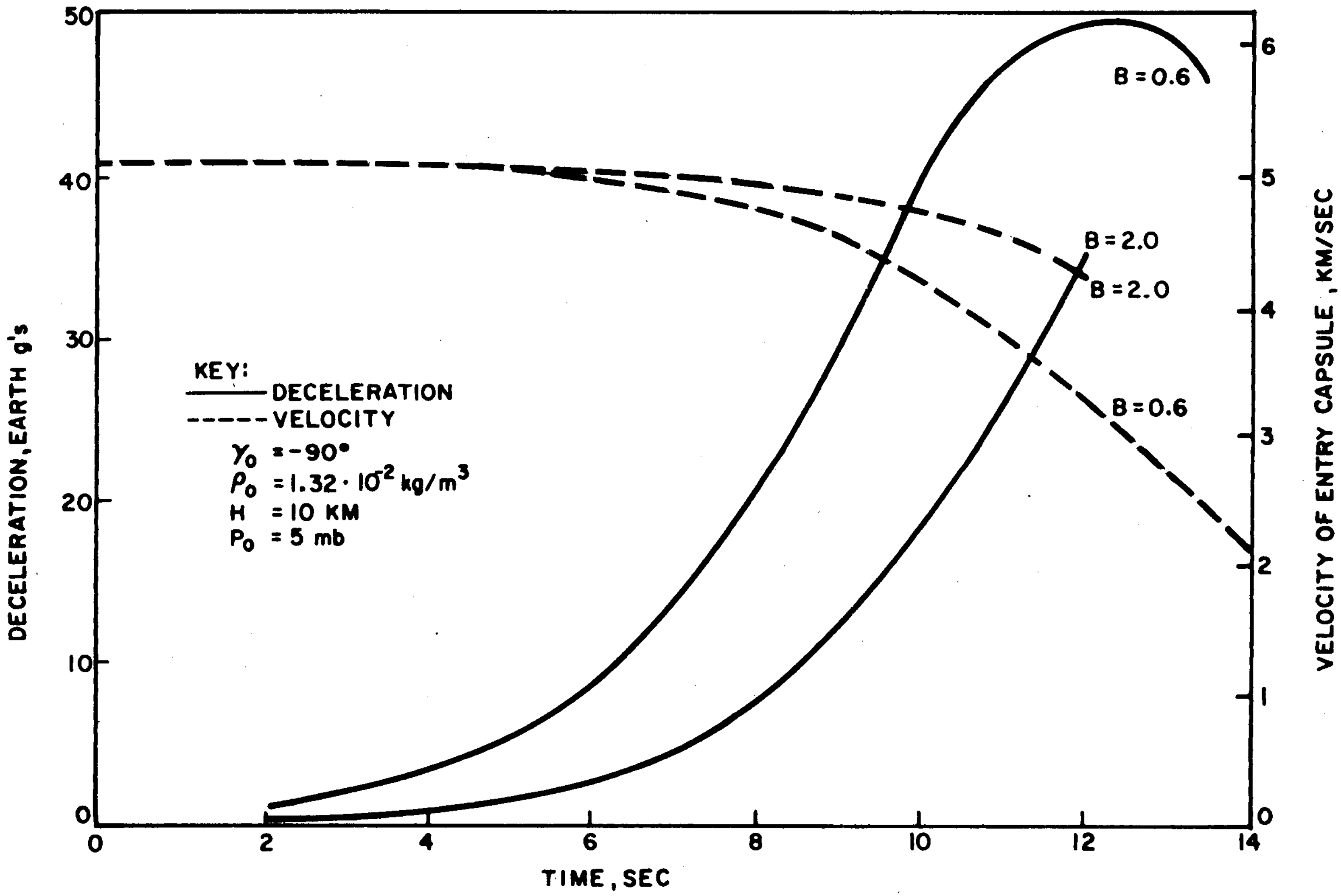


FIGURE A11. VELOCITY AND DECELERATION AS A FUNCTION OF TIME FOR DIRECT (γ_0 = -90°) ENTRY

Figures A12 through A15 show parametrically the way in which the terminal velocity of the entry vehicle depends on the atmospheric properties, the ballistic coefficient, the entry velocity, and the entry angle. For these figures the ignition velocity is defined for altitudes of 2.5, 5 and 10 km above the surface.

Figures A16 and A17 show that heating should not be a significant factor for either direct entry or descent from orbit.

In Figure A18, a curve of the residual velocity fraction, is presented to aid in comparison of our results with results derived from different atmospheric model assumptions. A broad line is used for the ratio (V/V_o) as the fractional rate of loss is about the same for the 3.8 and 7.1 km/sec entry velocities.

To patch the entry solution down to the surface of Mars with ideal soft landing conditions an impulse was calculated using an ISP = 310 sec and a structure factor f = 0.15. Tables A7 and A8 provide the calculated overall descent fractions for entry angles of 15° and 20°. Figure A19 is a plot of this data for the 15° entry case.

Figure A20 gives, for comparison, the residual weight fractions which have been calculated for Voyager. The curves refer to the stated entry weights.

Figure A21 provides a graphical description of the dependence of the entry problem on the aeroshell design. The

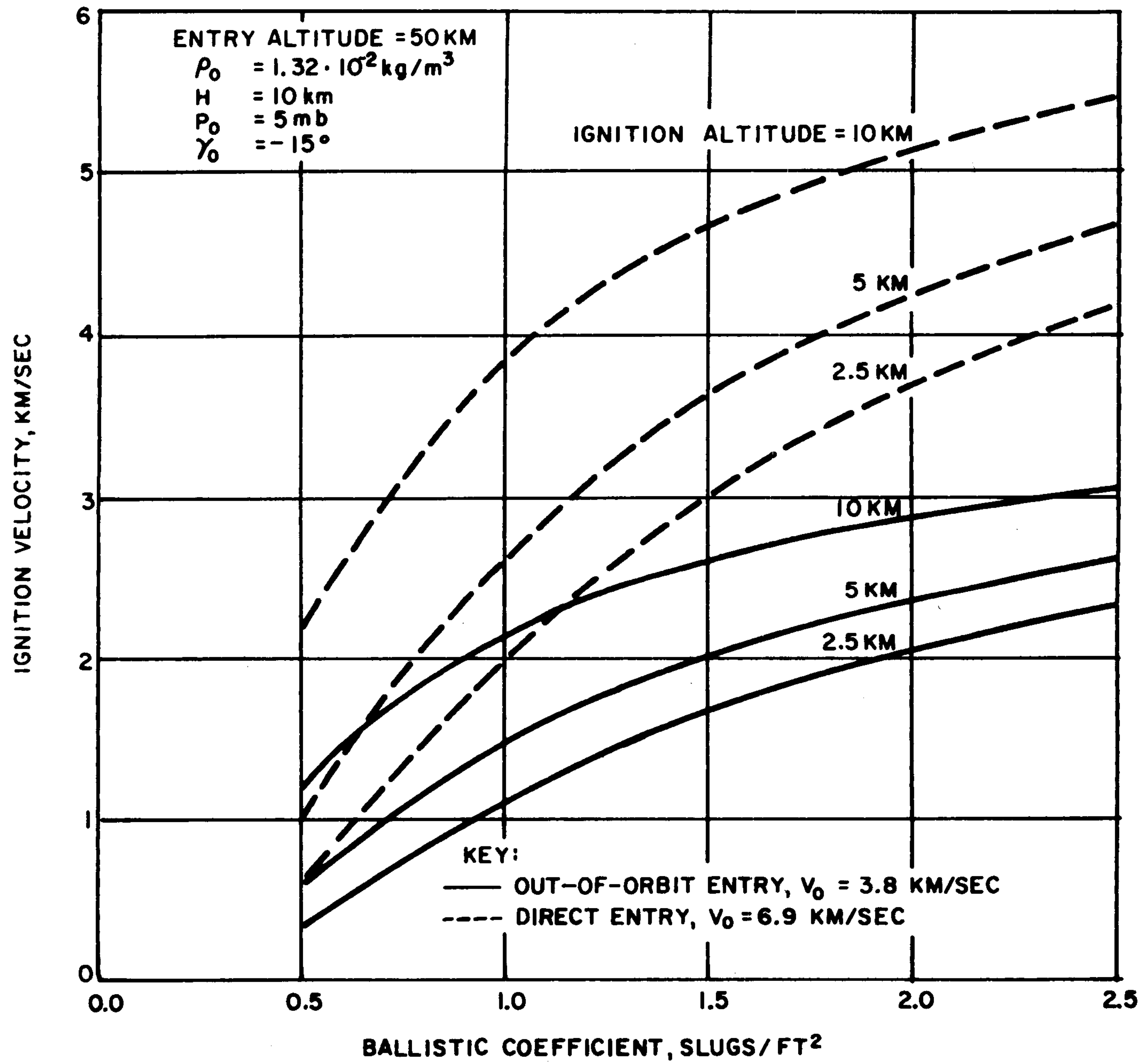


FIGURE A12. IGNITION VELOCITY AS A FUNCTION OF BALLISTIC COEFFICIENT (p_0 = 5 mb, γ = -15°)

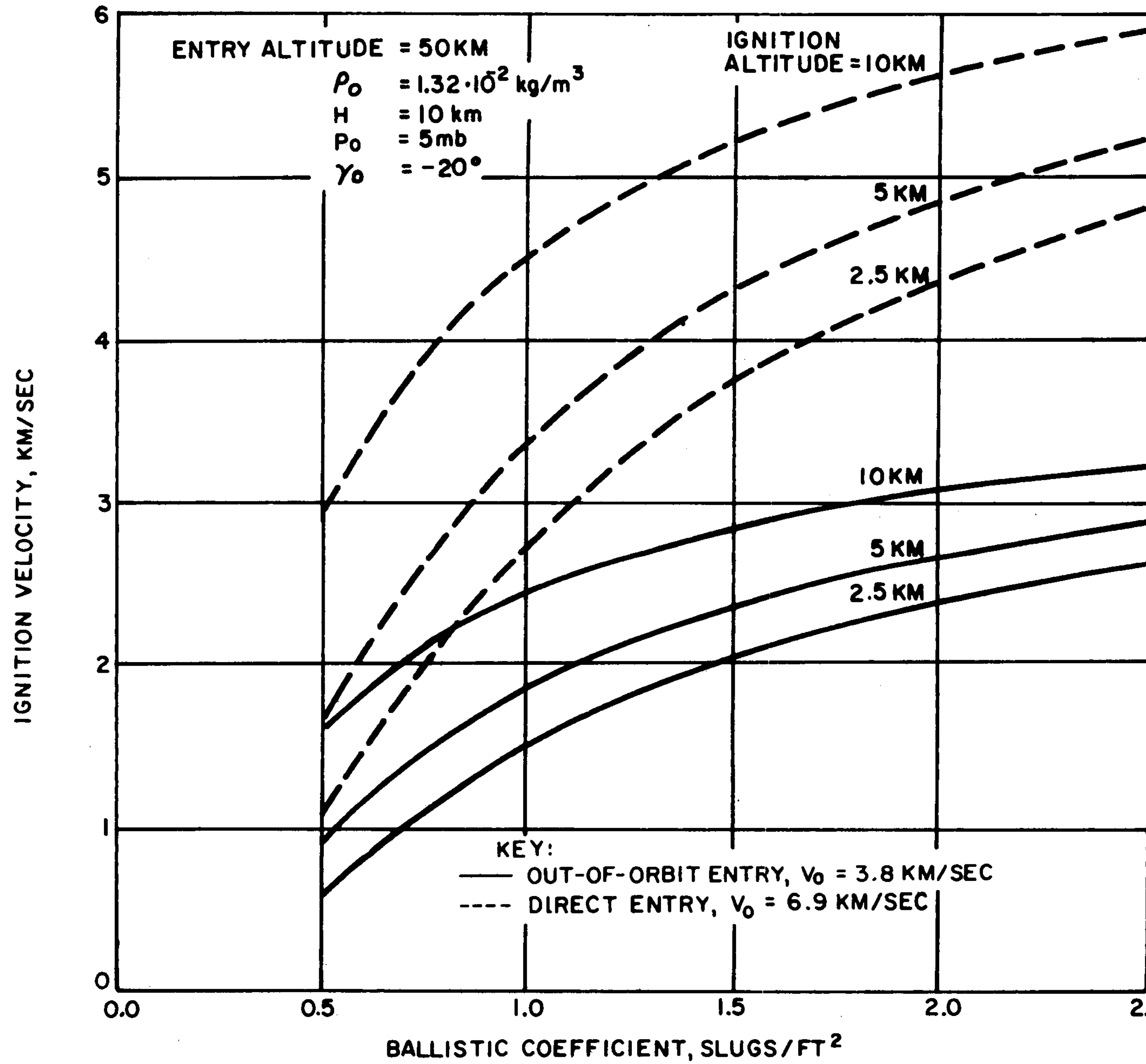


FIGURE A13. IGNITION VELOCITY AS A FUNCTION OF BALLISTIC COEFFICIENT
(p_o = 5mb, γ = -20°)

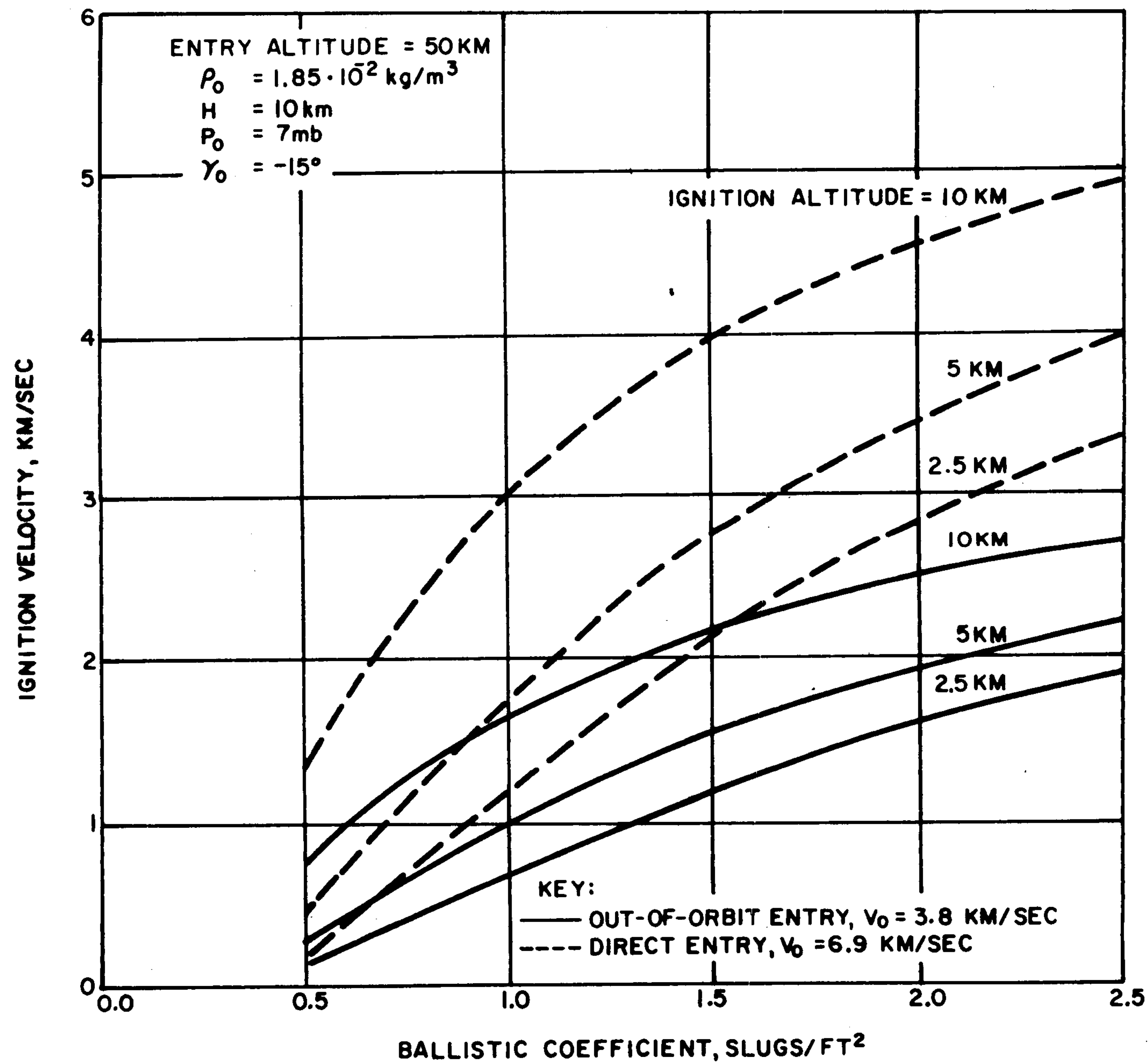


FIGURE A14. IGNITION VELOCITY AS A FUNCTION OF BALLISTIC COEFFICIENT
(p_0 = 7mb, γ = -15°)

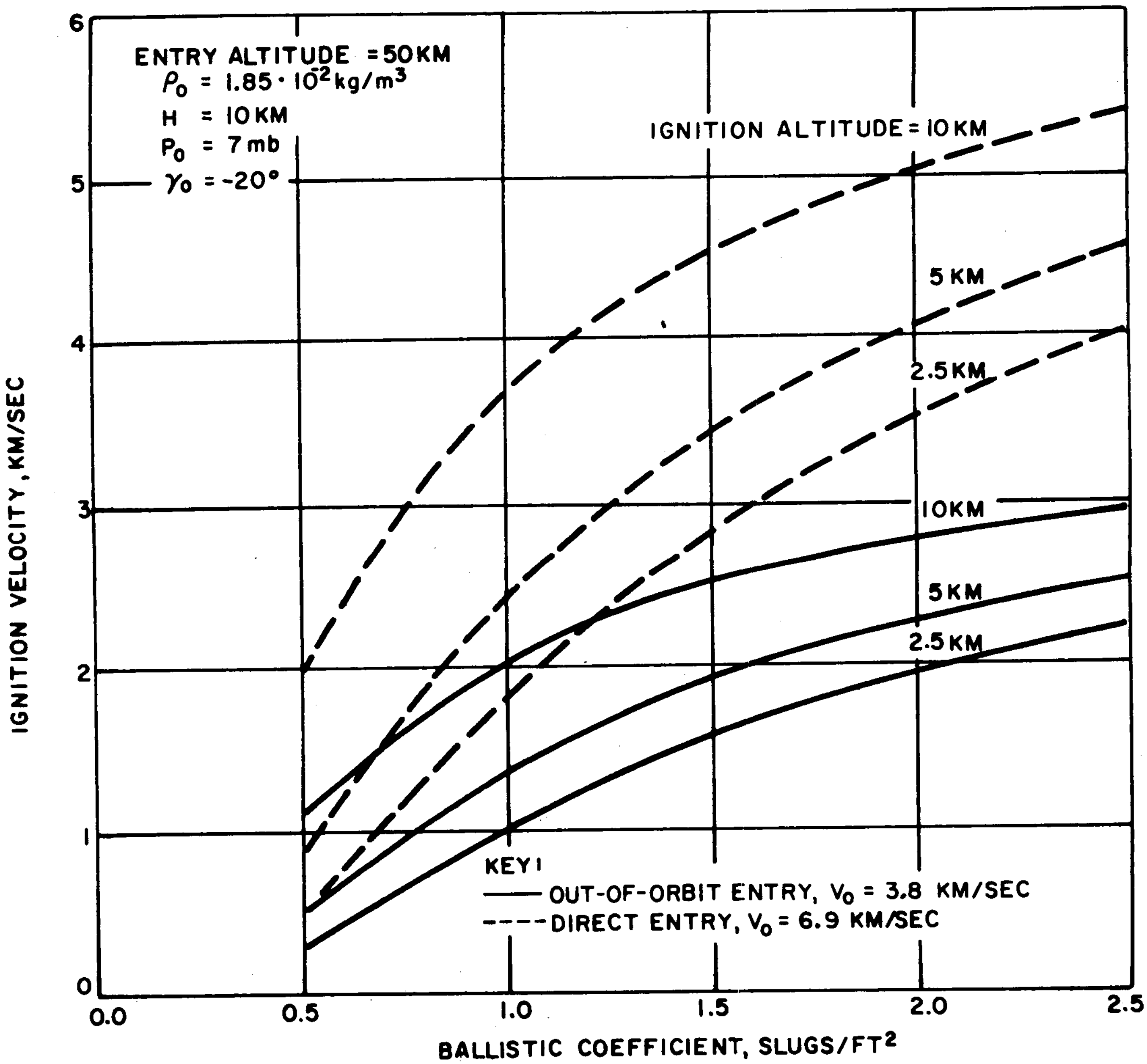


FIGURE A15. IGNITION VELOCITY AS A FUNCTION OF BALLISTIC COEFFICIENT (p_0 = 7mb, γ = −20°)

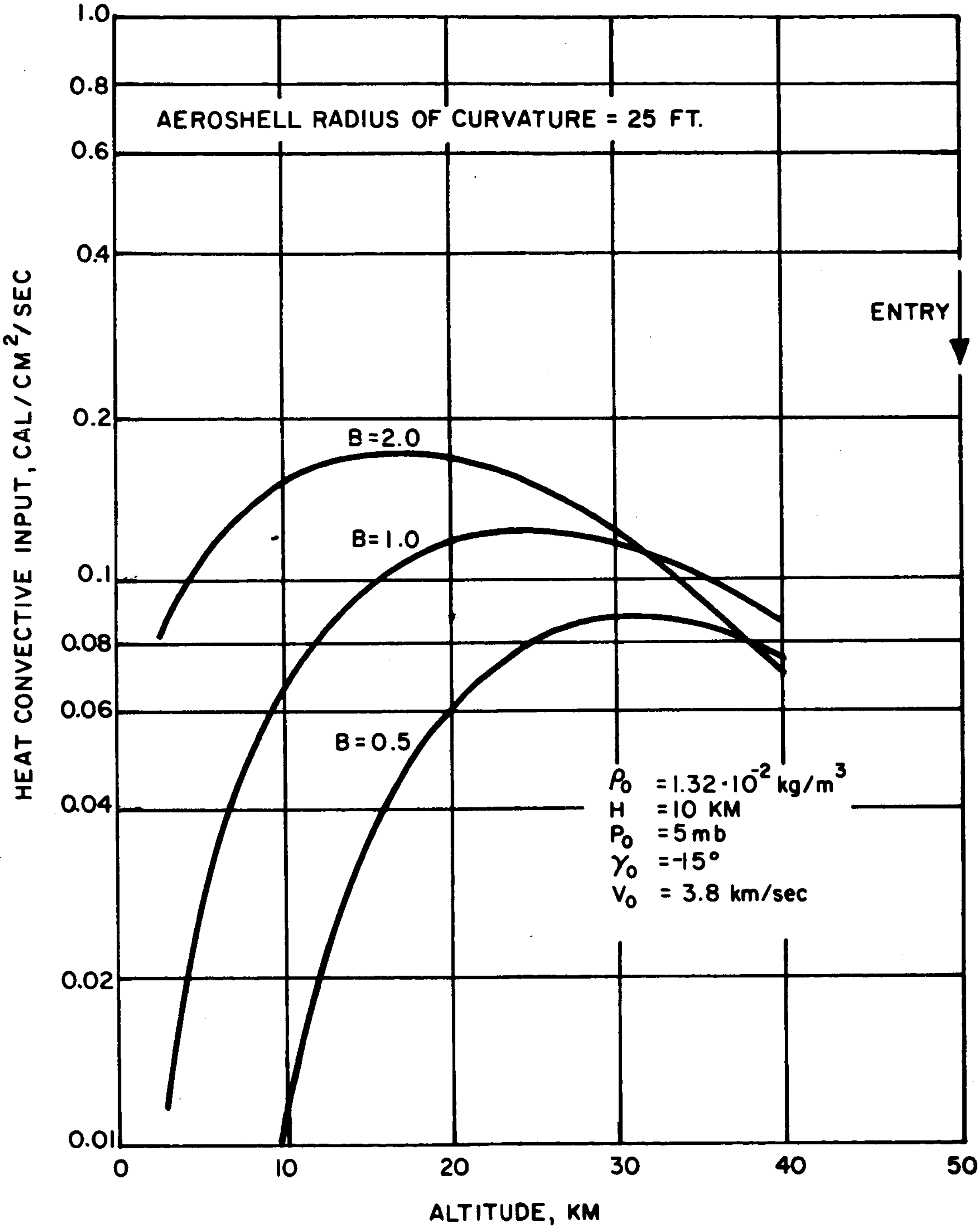


FIGURE A16. CONVECTIVE HEATING ESTIMATE FOR OUT-OF-ORBIT ENTRY

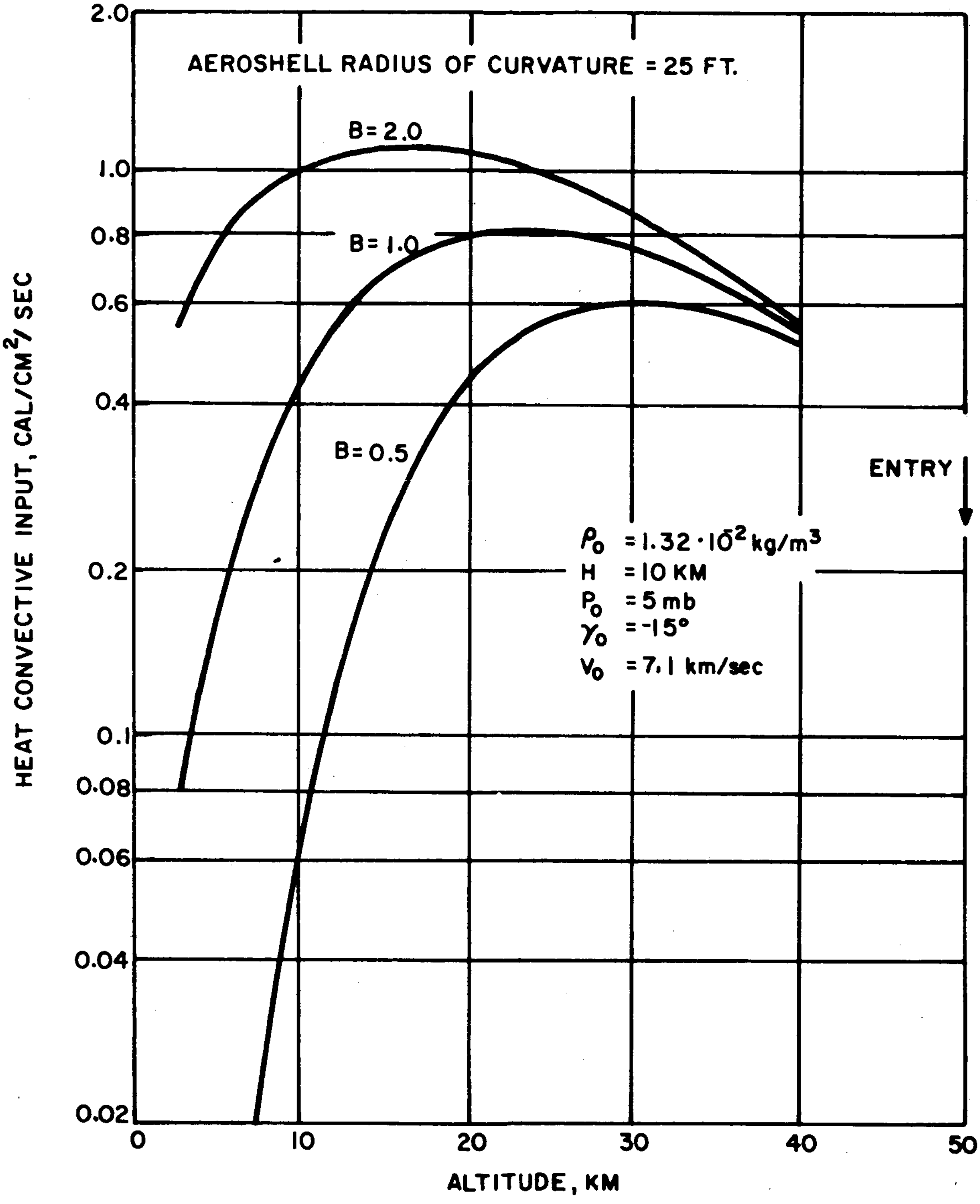


FIGURE A17. CONVECTIVE HEATING ESTIMATE FOR DIRECT ENTRY

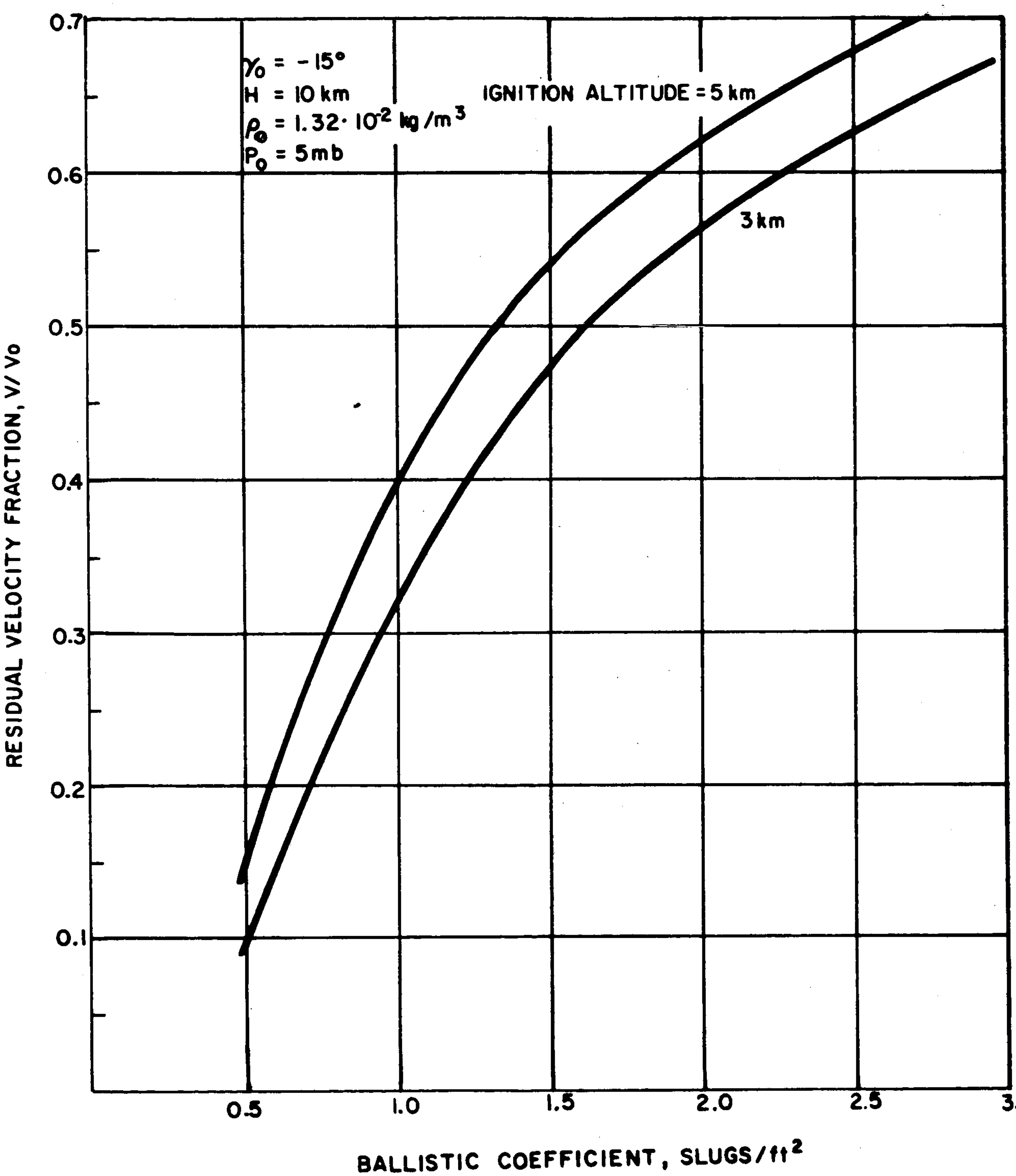


FIGURE A18. VELOCITY RESIDUAL AS A FUNCTION OF BALLISTIC COEFFICIENT

Table A-7

SUMMARY OF OVERALL DESCENT FRACTION

Entry Angle = -15°

B (slugs/ft^2)	V_o (km/sec)	h (km)	V_h (km/sec)	Descent Fraction*
0.5	3.8	3	0.37	0.61
0.5	3.8	5	0.56	0.56
0.5	7.1	3	0.66	0.54
0.5	7.1	5	0.83	0.47
0.8	3.8	3	0.86	0.50
0.8	3.8	5	1.14	0.45
0.8	7.1	3	1.56	0.37
0.8	7.1	5	2.10	0.30
1.0	3.8	3	1.17	0.44
1.0	3.8	5	1.45	0.39
1.0	7.1	3	2.15	0.29
1.0	7.1	5	2.68	0.23
1.5	3.8	3	1.74	0.35
1.5	3.8	5	2.01	0.31
1.5	7.1	3	3.20	0.13
1.5	7.1	5	3.70	0.09

*Descent fraction = $\frac{\text{landed weight}}{\text{entry weight}}$; ISP = 310 sec; structure factor f = 0.15; aeroshell weight fraction = 0.3.

Table A-8

SUMMARY OF OVERALL DESCENT FRACTION

Entry Angle = -20°

B (slugs/ft^2)	V_o (km/sec)	h (km)	V_h (km/sec)	Descent Fraction*
0.5	3.8	3	0.64	0.55
0.5	3.8	5	0.72	0.50
0.5	7.1	3	1.17	0.44
0.5	7.1	5	1.62	0.37
0.8	3.8	3	1.25	0.43
0.8	3.8	5	1.53	0.38
0.8	7.1	3	2.30	0.27
0.8	7.1	5	2.82	0.21
1.0	3.8	3	1.56	0.38
1.0	3.8	5	1.84	0.33
1.0	7.1	3	2.88	0.21
1.0	7.1	5	3.40	0.16
1.5	3.8	3	2.11	0.30
1.5	3.8	5	2.35	0.27
1.5	7.1	3	3.89	0.12
1.5	7.1	5	4.35	0.09

*Descent fraction = $\frac{\text{landed weight}}{\text{entry weight}}$; ISP = 310 sec; structure factor f = 0.15; aeroshell weight fraction = 0.3.

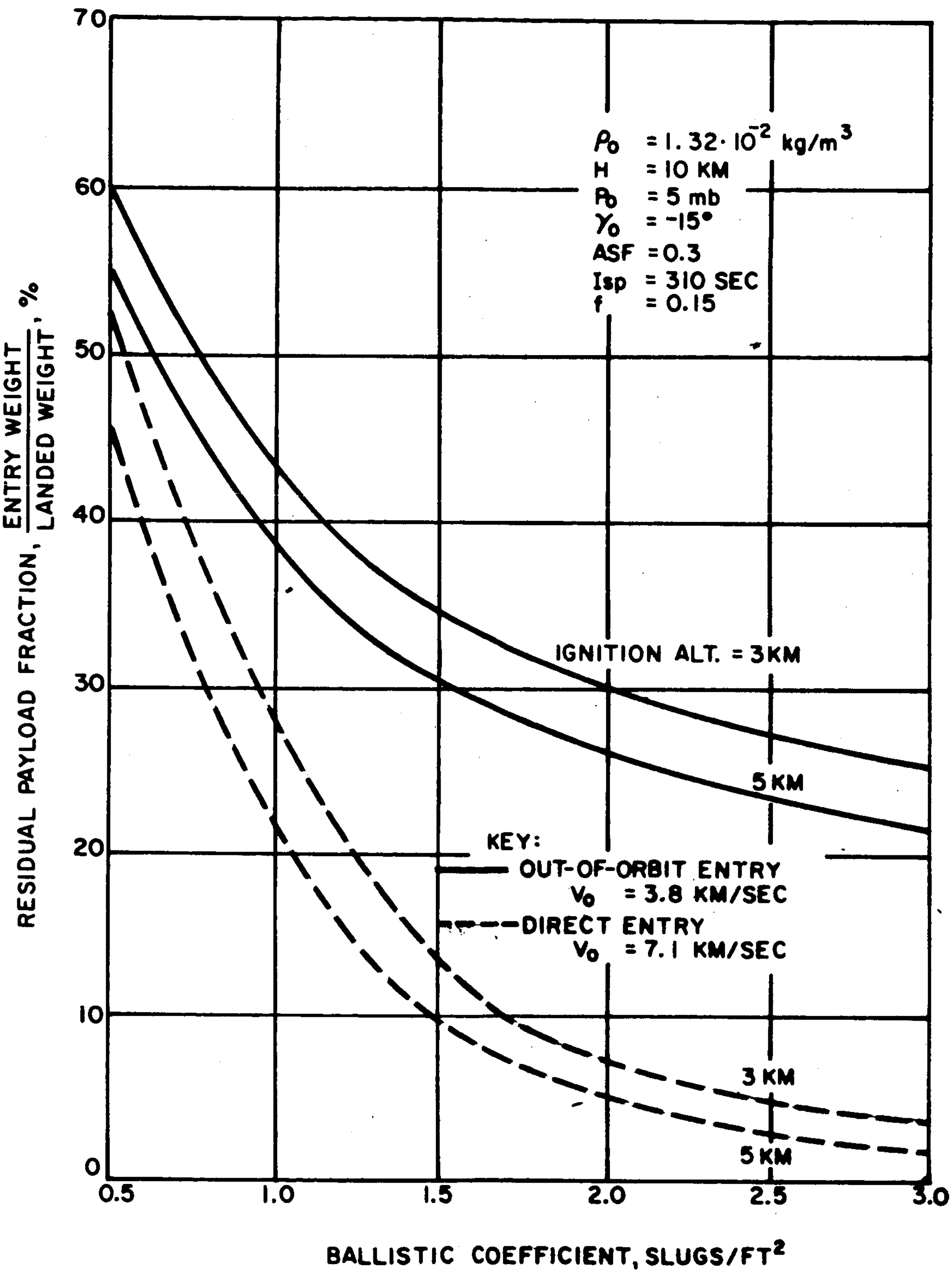


FIGURE A19. RESIDUAL WEIGHT VS. BALLISTIC COEFFICIENT

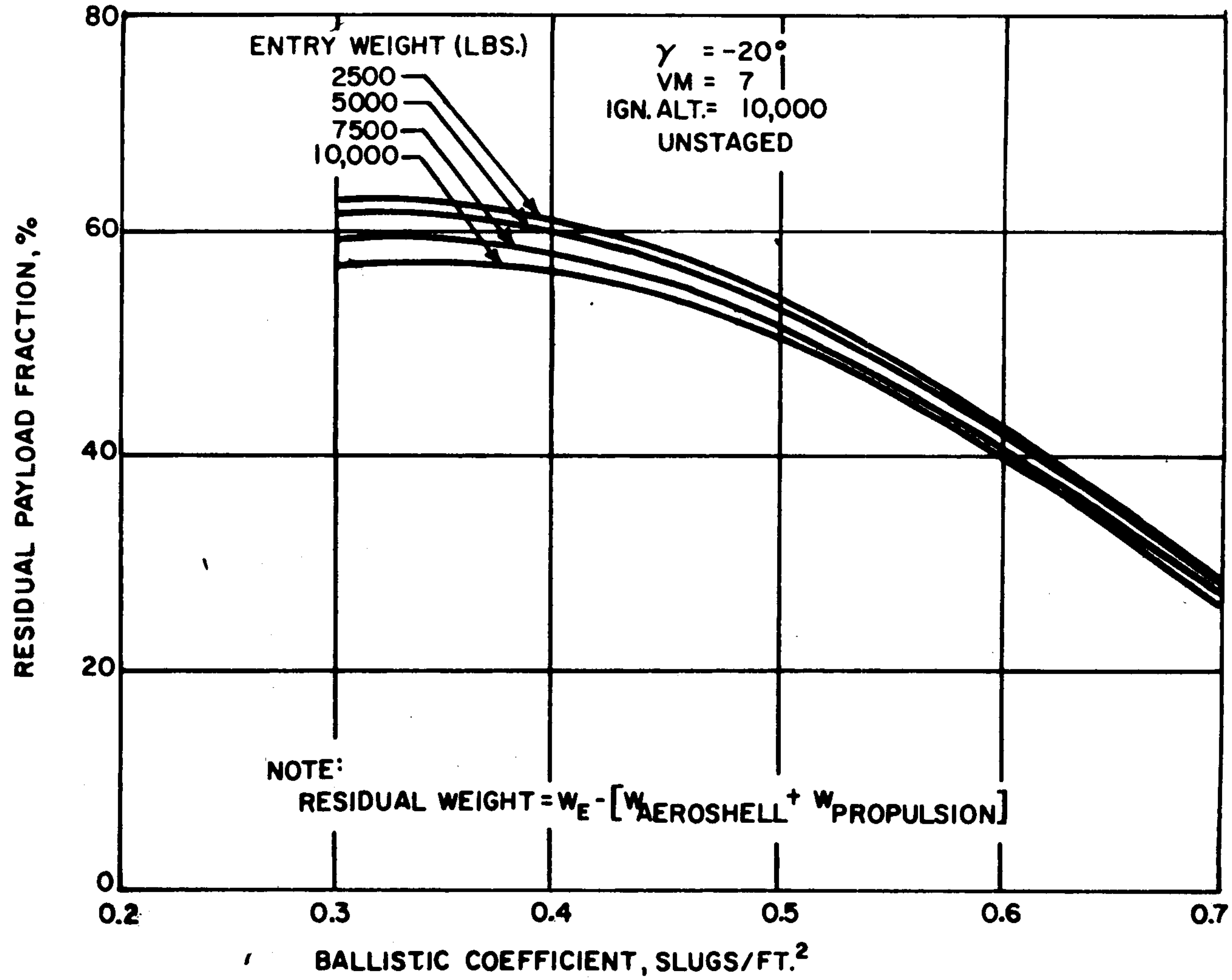


FIGURE A20. RESIDUAL WEIGHT VS. BALLISTIC COEFFICIENT FROM JPL VOYAGER STUDY

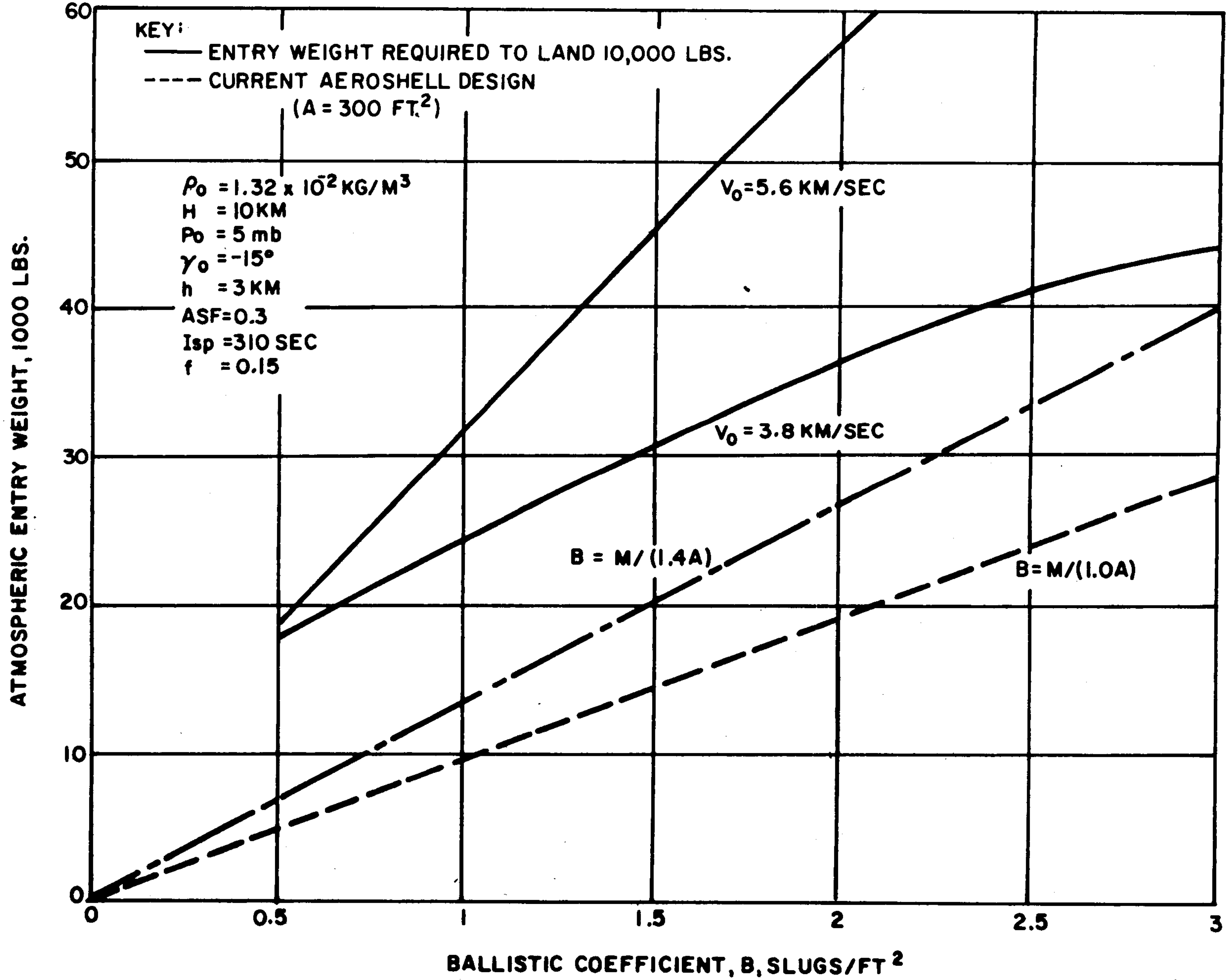


FIGURE A21. ENTRY WEIGHT REQUIREMENT AND CURRENT AEROSHELL DESIGN COMPARISON full line curves show the entry weight required to land 10,000 lb on the surface of Mars using the data from Table A7. The dashed curves show the current state of the art of aeroshell design with a diameter limitation of 20 ft (i.e., 300 ft^2 area). Advanced technological development will be required to achieve ballistic coefficients in the 0.5 to 1.0 range within this aeroshell diameter limitation for heavy Mars landers. A possible design difficulty with aeroshell configurations which must shield massive entry weights should also be noted. The center of gravity of the entry craft could lay well outside the shell-cover, thereby creating a serious problem of stability through the generation of torques.

From the preceding figures and tables we have inferred that serious design problems will probably exist for landed weights in excess of 5000 lb under any circumstance. We have therefore briefly investigated the advantages of going to lifting reentry and/or skip. Two such profiles are shown in Figures A22 and A23. The net effect of either lift or skip is to measurably increase the deceleration path through the sensible atmosphere. On the basis of our limited investigation, it seems to offer attractive alternatives for large values of ballistic coefficient. Table A9 contains trajectory summaries for a range of L/D ratios.

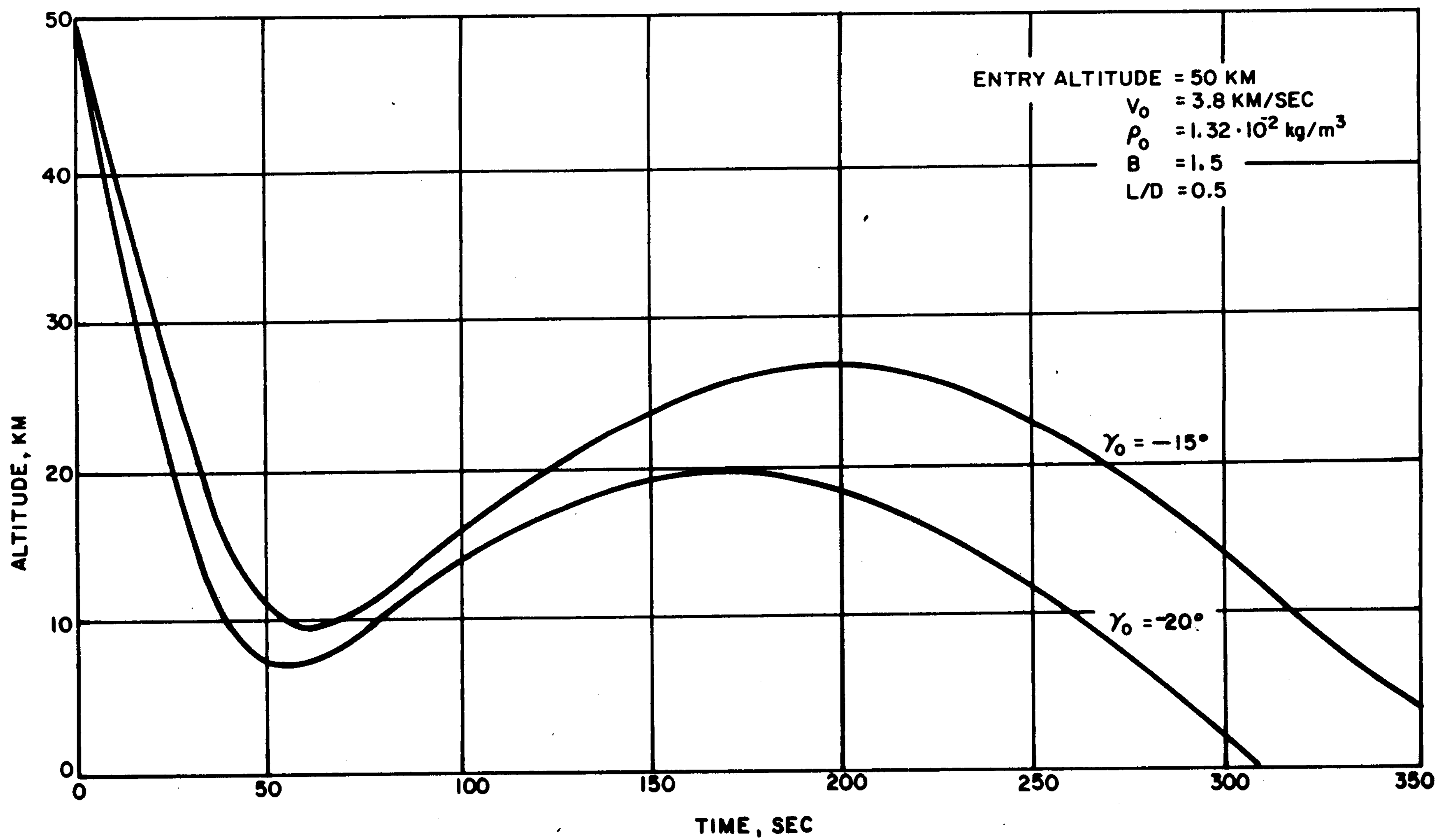


FIGURE A22. ALTITUDE PROFILES FOR L/D = 0.5

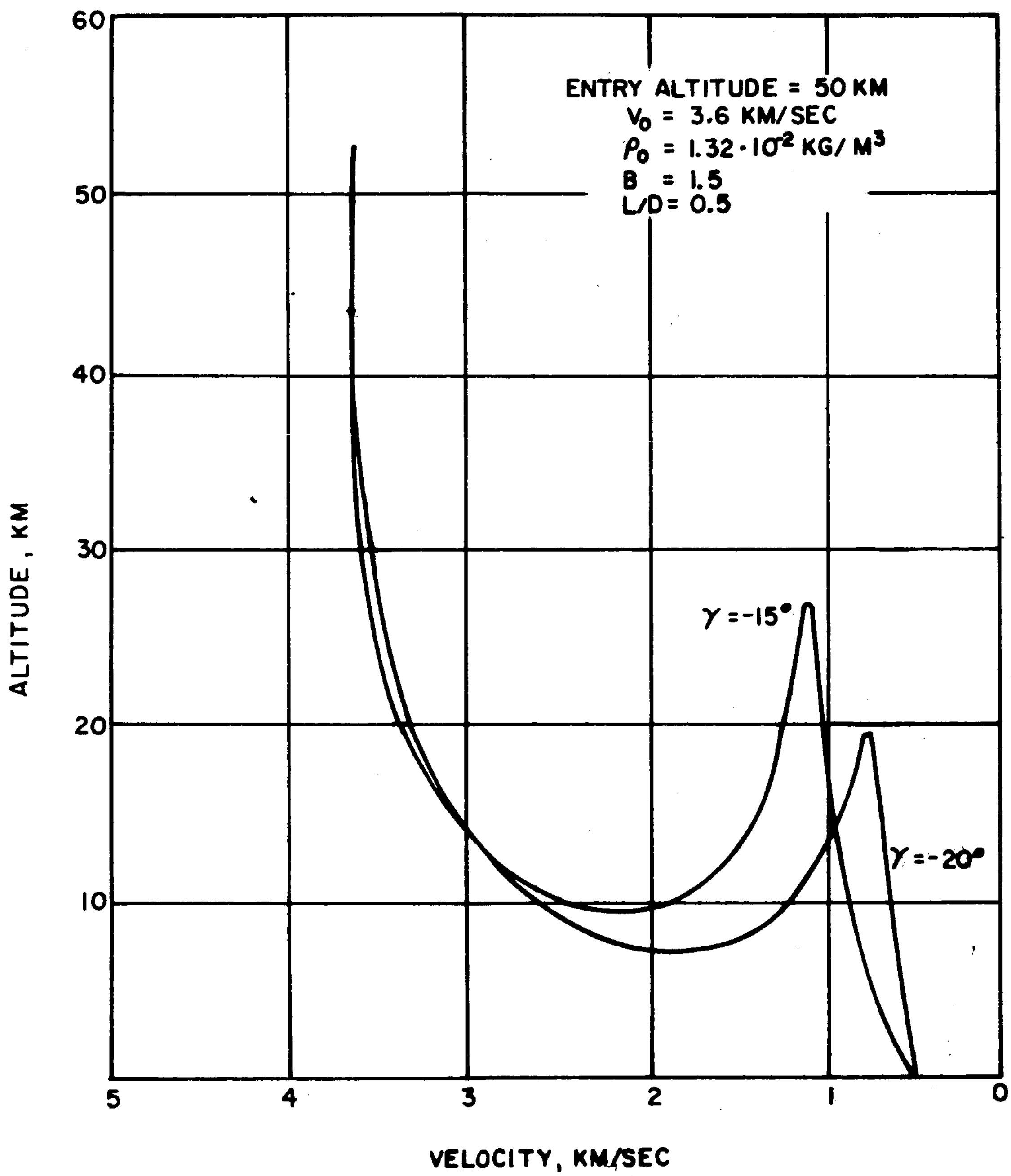


FIGURE A23. VELOCITY VS. ALTITUDE FOR L/D=0.5

Table A-9

L/D EFFECTS FOR EXTREME VALUES OF ENTRY VELOCITY

L/D	B (slugs/ft^2)	V_o (km/sec)	Time of Flight (sec)	Impact Velocity (km/sec)	Skip-out* Velocity (km/sec)	Final Path Angle
0.0	0.5	12.5	108	0.29	-	-29.5
0.0	1.0	12.5	70	0.94	-	-10.9
0.5	0.5	12.5	88	-	6.20	14.5
0.5	1.0	12.5	94	-	6.45	14.5
0.0	0.5	6.25	132	0.37	-	-30.1
0.0	1.0	6.25	100	0.88	-	-14.5
0.5	0.5	6.25	224	-	2.54	8.86
0.5	1.0	6.25	242	-	2.61	8.63
0.0	0.5	3.8	150	0.33	-	-31.1
0.0	1.0	3.8	124	0.87	-	-18.9
0.5	0.5	3.8	374	0.77	-	-15.8
0.5	1.0	3.8	374	0.80	-	-15.2

*Skip-out is assumed at 100 km altitude.

Initial Conditions

$\gamma = -15°$
H = 10 km
Entry altitude = 100 km

$\rho_o = 1.32 \times 10^{-2}$ kg/m^3
$P_o = 5$ mb

A.5 MARS LAUNCH REQUIREMENTS

Lewis Research Center provided Mars launch vehicle payload ratios for the Mars launch phase of the mission. Their analysis was performed with a calculus of variations, multistage launch vehicle computer program. A description of the variational method used has been published (Teren and Spurlock 1966).

Launch and ascent data associated with the Mars ascent payload rations and used in the mission mode weight statements are listed in Table A-10. The most important numbers to note in the table are the low thrust-to-weight ratios of the second stage. These values imply extremely long burning times for the second stage (as high as 15 minutes). Higher second stage thrust-to-weight ratios (0.5 to 0.8) were investigated as well. However, in order to match the performance of the cases shown in Table A-10 coast times approaching 1000 seconds between first and second stage burn are required. No attempt was made to decide whether high or low thrust second stage design is the more practical. The performance of both are about the same. Further analysis will be required to resolve which design is better.

A.6 RENDEZVOUS AND DOCKING

The feasibility of both automated rendezvous and docking in Mars orbit must be established from both a systems design and an energy viewpoint. For rendezvous, the more important point is probably the energy requirements while for docking,

Table A-10

MARS LAUNCH PHASE DATA
(Computed by Lewis Research Center)

Parameters	Case 1*	Case 2**
Liftoff weight	10,000	1,000
First stage		
Thrust (lb)	10,000	1,000
Thrust-to-weight ratio	1	1
ISP (sec)	310	310
Structure factor - f (ASC)	0.15	0.15
σ (JPL)	0.13	0.13
Propellant loading (lb)	6,489	640
Hardware weight (lb)	973	96
Altitude of variational steering start	100,062'	97,000'
Burnout altitude (ft)	219,000	238,000
Second stage		
Initial weight (lb)	2,538	264
Thrust (lb)	200	30
Thrust-to-weight ratio	0.0788	0.114
ISP (sec)	310	310
Structure factor - f (ASC)	0.20	0.20
σ (JPL)	0.167	0.167
Propellant loading	665	98
Hardware weight	133	20
Orbit (Mars radii)	1.1 x 1.1	1.3 x 1.3
Orbit payload (lb)	1,740	146
Liftoff weight: orbit payload ratio	5.75:1	6.85:1
Ideal velocities		
First stage	10,400	10,180
Second stage	3,000	4,514

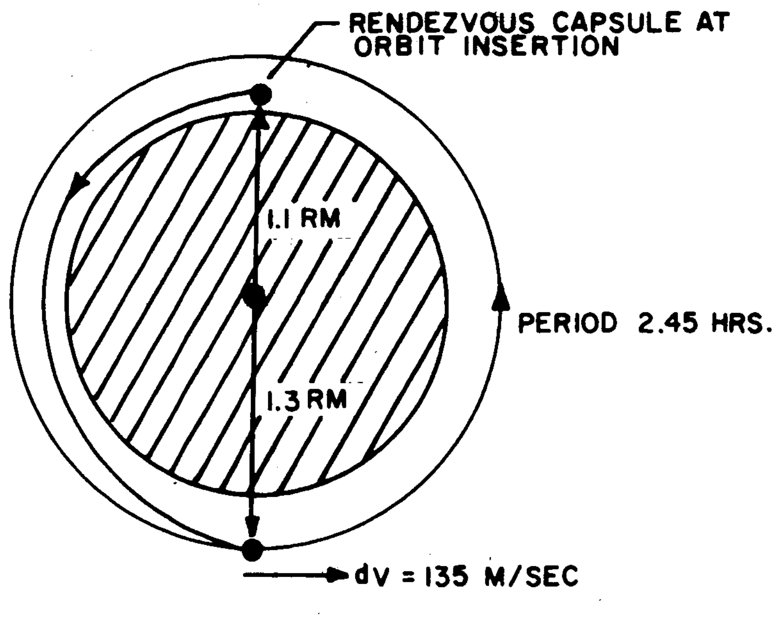
*Case 1 is for launch to parking orbit (1.1 Mars radii circular).
**Case 2 is for launch to rendezvous (1.3 Mars radii circular).

the system design problems are paramount. This section is restricted to consideration of the energy requirements for a particular automated rendezvous scheme in Mars orbit.

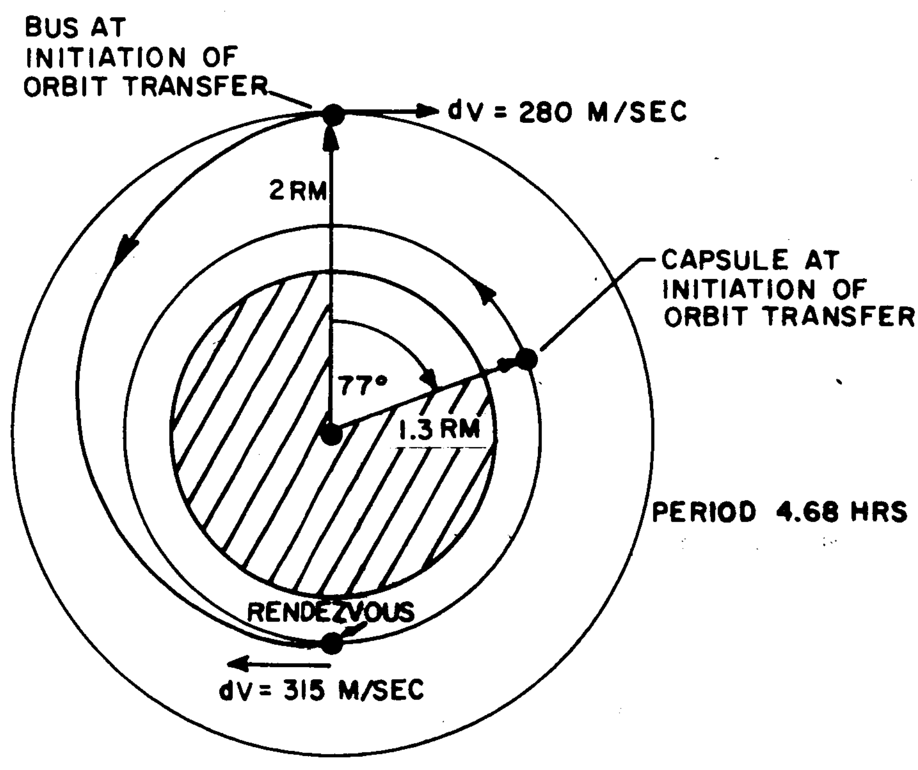
The nominal conditions and requirements of the orbital transfer and rendezvous maneuvers are illustrated in Figure A24. The rendezvous capsule is launched from the Martian surface upon command from either the Earth or the rendezvous bus, and is inserted at periapse into a 1.1 x 1.3 Mars radii (R_M) orbit. Upon reaching the apoapse point 1.1 hrs after insertion, the capsule is placed into a circular orbit of radius 1.3 R_M at a dV expenditure of 135 m/sec.

The remaining phases of the orbital transfer and rendezvous maneuver are accomplished by the rendezvous bus which is initially in a circular orbit of radius 2 R_M. A Hohmann transfer is initiated when the bus leads the capsule by an angle approximately equal to 77°. The nominal dV cost of the 2-impulse transfer is 280 + 315 = 595 m/sec. Actually, the second impulse is to be considered part of the terminal phase of rendezvous as will be discussed shortly.

Initiation of the Hohmann transfer will be made only after the orbit of the rendezvous bus has been well determined by Earth-based tracking over several orbits. The synodic period of the bus and capsule during this waiting interval is 5.15 hrs. Since this corresponds to slightly more than two orbital periods of the rendezvous capsule, a waiting interval



a.) ORBITAL TRANSFER MANEUVER OF CAPSULE



b.) ORBITAL TRANSFER MANEUVERS OF BUS

FIGURE A24. NOMINAL ORBITAL TRANSFER AND RENDEZVOUS of about 10 hrs would seem a reasonable requirement for accurate orbit determination of the capsule. Figure A25 shows the separation distance between the capsule and bus prior to initiation of the orbital transfer maneuver. It may be desirable to maintain a communication link between capsule and bus during this time - possibly for the transfer of information from and to Earth. It is to be noted that this communication link is only open about 56 percent of the time due to the capsule-bus occultation by Mars.

Due to launch azimuth errors or timing constraints, the orbital planes of the rendezvous capsule and bus will not be matched exactly. It is expected that the orbital plane adjustment will be performed by the bus as part of the transfer maneuver. Hence, assuming a maximum inclination error of 2°, the plane change requirement is 90 m/sec. With the addition of a 55 m/sec pad for excess maneuvering during the terminal and docking phases, the total dV requirement for the bus is 740 m/sec.

The terminal phase of rendezvous and docking is performed by the bus and capsule systems independent of Earth tracking and command. The thrust maneuvers are made by the bus which plays the role of the interceptor vehicle. One possible configuration would have fixed direction thrustors placed along the longitudinal and normal axes and operating in an on-off mode. Two thrust levels may be required for efficient maneuvering in the presence of initial condition and system instrumentation

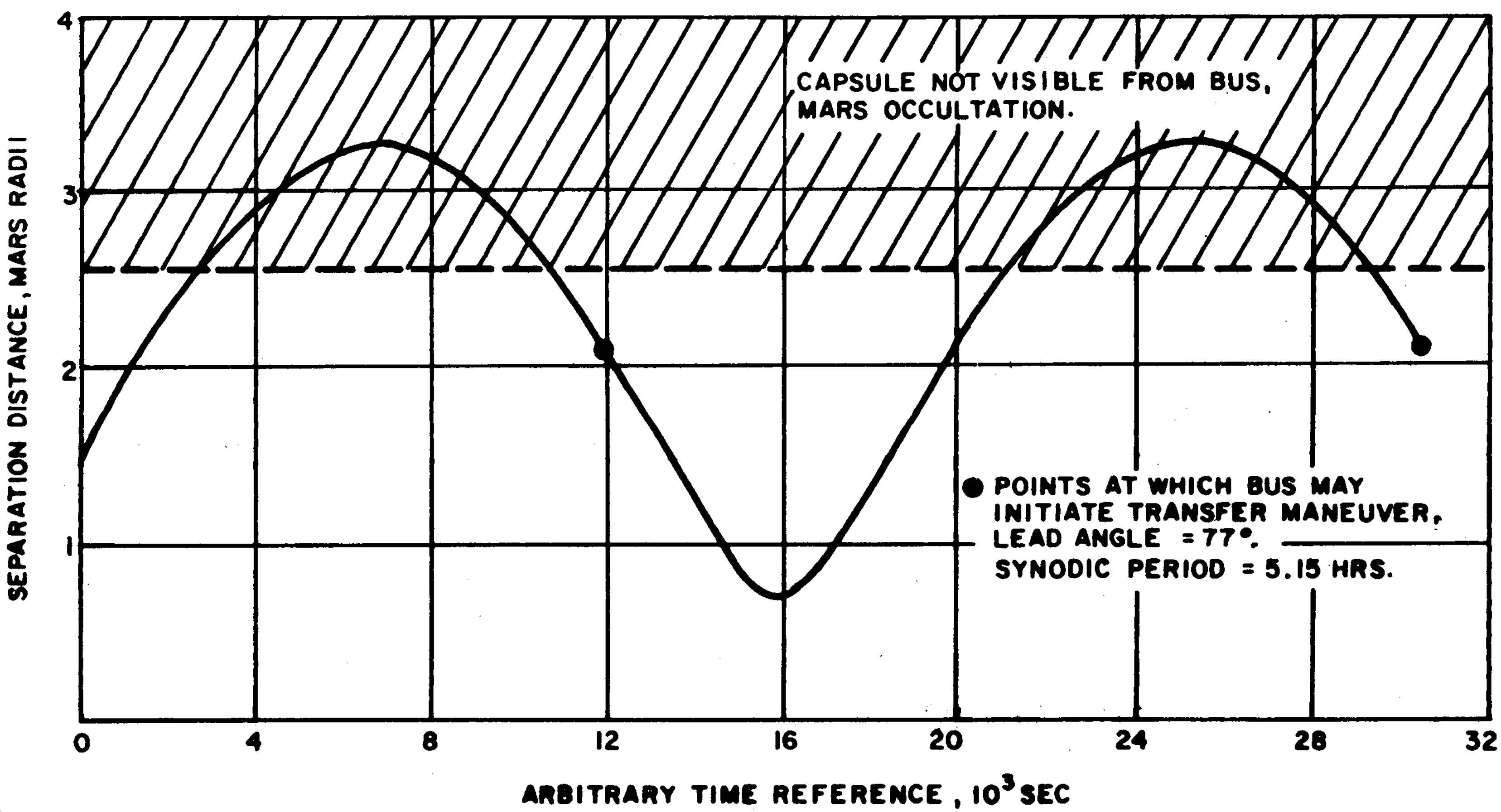
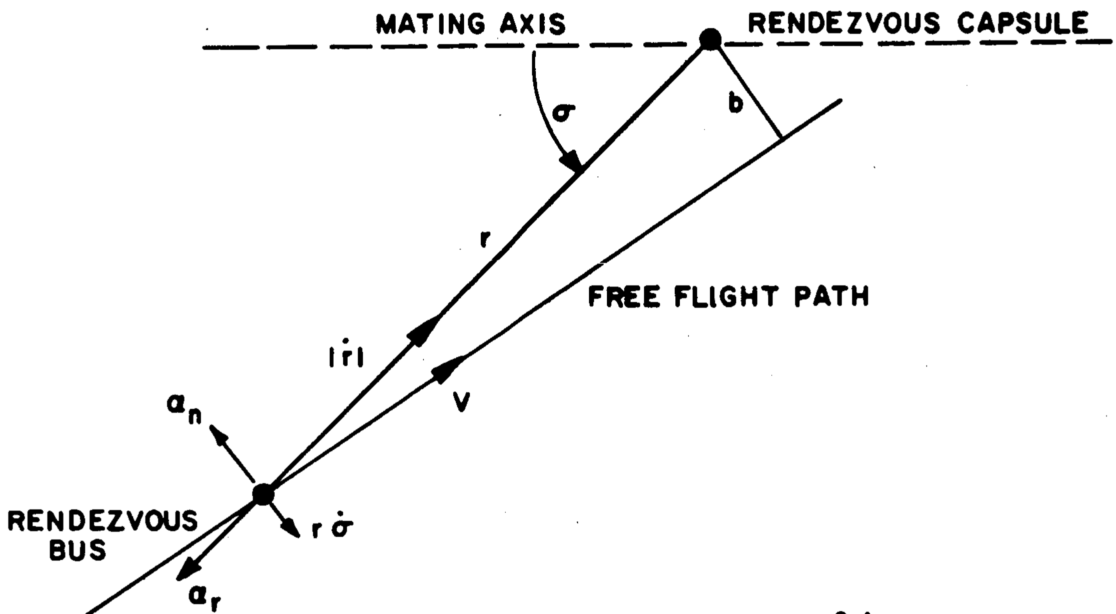


FIGURE A25. SEPARATION DISTANCE BETWEEN RENDEZVOUS CAPSULE AND BUS PRIOR TO INITIATION OF TRANSFER MANEUVER errors. The main function of the capsule (target vehicle) during the terminal phase will be to maintain the proper attitude of its mating axis as commanded by the bus. Since the approach direction to the target may vary considerably from the nominal direction, the maneuvering requirements of the bus during the terminal phase may be minimized by placing the burden of mating axis alignment on the capsule.

Figure A26 illustrates the simplified geometry of the terminal rendezvous maneuver. A relative coordinate system is established by the rendezvous radar and inertial gyro references contained on-board the bus. The variables to be measured are the range (r), range-rate ($\dot{r}$) and the line-of-sight rate ($\dot{\sigma}$). In addition, a measurement of the mating axis misalignment (σ) will be required in order to command the proper capsule attitude. The terminal phase is initiated at the radar acquisition range of about 50 km. Assuming an on-off thruster mode, the velocity corrections are divided into two orthogonal channels. The first channel uses range-rate or longitudinal information along the line-of-sight direction functions to reduce the closing velocity to a small value consistent with the docking requirements. The second channel, normal to the line-of-sight, functions to null the miss distance (b). Channel orthogonality is maintained by the attitude control system of the bus which nulls the radar dish gimbal angle thus driving the longitudinal and radar axes into coincidence.



$$b = \frac{r^2\dot{\sigma}}{(\dot{r}^2 + r^2\dot{\sigma}^2)^{1/2}}$$

$$= \frac{r^2\dot{\sigma}}{|\dot{r}|} \qquad \underline{r \gg b}$$

FIGURE A26. TERMINAL RENDEZVOUS GEOMETRY

One possible form of the on-off control logic for each channel is listed on Table A-11. The switching criteria of the range-rate channel is designed to achieve rendezvous in the shortest time consistent with the available thrust acceleration, unilateral thrust direction, and the desire to minimize the frequency of normal corrections. Time-to-rendezvous is maintained between the τ_{min} and τ_{max}. The interval $(\tau_{max} - \tau_{min})$ between longitudinal thrusting is available for the measurement, determination and actuation of normal velocity corrections. Behavior of the range-rate control system is illustrated by the phase plane plot in Figure A27 which assumes an initial closing rate of 316 m/sec, a longitudinal acceleration of 2 m/sec^2 (0.2 g), a minimum time-to-rendezvous of 40 seconds, and a 20 second interval between corrections. Since the thrust-on time reduces to very small values (approaching the thruster response time), it would be desirable to switch to a smaller thrust level during the last few hundred meters of the rendezvous maneuver.

The control logic of the normal velocity channel calls for thrust initiation when the line-of-sight rate exceeds some prespecified threshold value. Thrust is continued until the integrating accelerometer in the normal axis indicates that the computed velocity increment has been applied. Since a finite time is required to build up this increment, there will be a residual line-of-sight rate remaining at thrust cutoff. This residual rate will then build up due to the coupling between the longitudinal and normal velocity channels. Hence, a limit

Table A-11

TERMINAL RENDEZVOUS AND DOCKING LOGIC

MEASURABLES

Range, r

Range rate, $\dot{r}$

Loss rate, $\dot{\sigma}$

Mating axis angle, σ (docking)

CONTROL

Radial acceleration a_r

Normal acceleration a_n

TERMINAL MANEUVER (ON-OFF CONTROL)

1. Range rate channel

Thrust on: a) $r = \tau_{min} (-\dot{r})$

$$\text{if } r \geqslant \frac{\dot{r}^2}{a_r}$$

b) $r = \frac{\dot{r}^2}{2a_r} + \frac{a_r}{2} \quad \tau_{min}{}^2$

$$\text{if } r < \frac{\dot{r}^2}{a_r}$$

Thrust off: $\frac{r}{-\dot{r}} = \tau_{max}$

2. Normal Velocity Channel

Thrust on: $\dot{\sigma} \geqslant \dot{\sigma}_{max}$

Thrust off: $V_n = a_n t = r\dot{\sigma}$ (at thrust on time)

DOCKING MANEUVER

1. Maintain $|\dot{r}| \leqslant \dot{r}_{max}$
2. Maintain $|\dot{\sigma}| \leqslant \frac{\dot{r}}{r^2} \, b_{max}$
3. Turn rendezvous capsule so that $\sigma \leqslant \sigma_{max}$

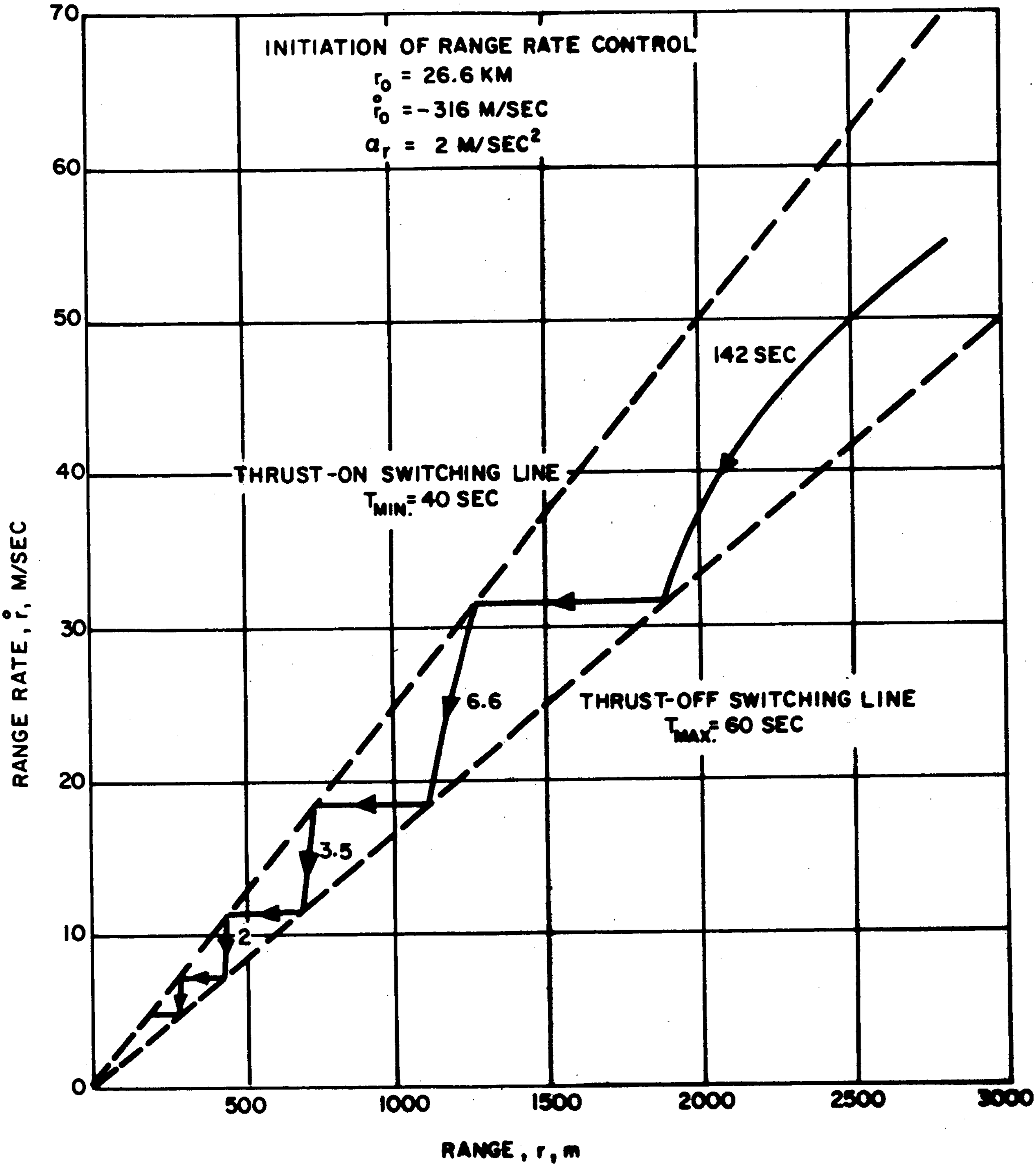


FIGURE A27. RANGE RATE CONTROL CHANNEL-TERMINAL PHASE cycle is established in that the normal velocity corrections are unilateral in direction, thus eliminating the necessity of rolling the vehicle between normal corrections.

A.7 EARTH REENTRY

It has been assumed that it is feasible to contain the 1-2 lb Mars sample within a 50-55 lb Earth reentry capsule (JPL 1967). Two options were considered for the returning Mars sample as it approaches the Earth. These were direct atmospheric entry and injection into an Earth orbit.

It was assumed for direct Earth reentry speeds up to 40,000 ft/sec that almost vertical entry can be made. This will minimize range dispersion errors and the heat conduction to the sample due to the short deceleration period. The method of sample retrieval can be either by air snatch, if the range dispersion error is low, or by ground or sea recovery. In either event a final descent by parachute would be required.

An alternative mode of returning the Mars sample is to inject it into Earth orbit and to retrieve it using an additional (possibly manned) Earth orbital mission. Atmospheric entry speed at Earth is shown in Figure A28 as a function of the hyperbolic approach velocity (VHP). To maintain entry velocities below 40,000 ft the VHP must be constrained to less than 5 km/sec. For approach velocities in excess of this, some form of retro propulsion will be required. However it is not clear whether it is better to use this retro propulsion to inject the capsule into orbit or to slow it down to 40,000 ft/sec

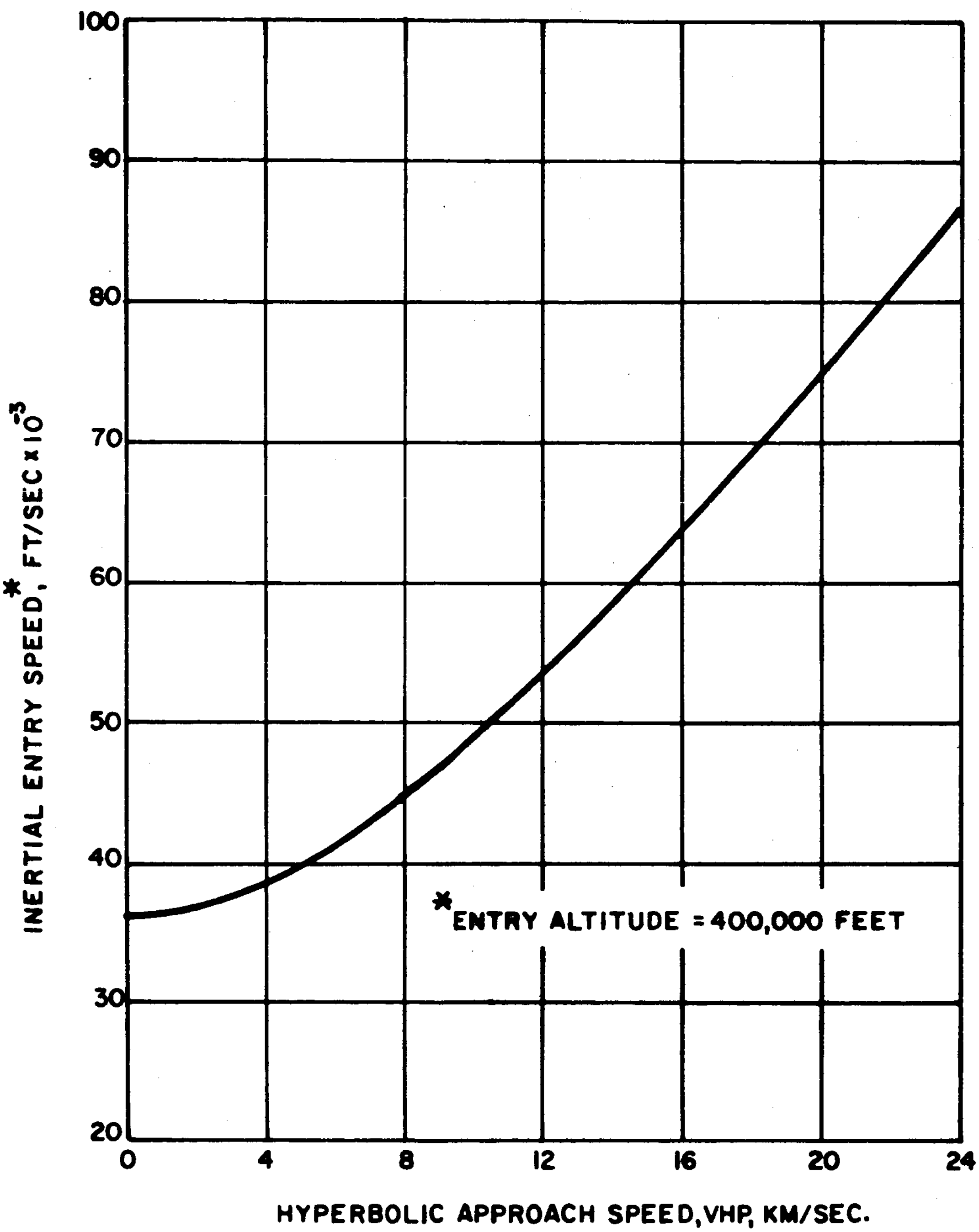


FIGURE A28. EARTH ATMOSPHERIC ENTRY SPEED FOR DIRECT DESCENT.

for direct entry. Purely from an energy standpoint, direct entry would always seem to be simpler and less expensive, but system design considerations may oppose this.

Since in evaluating the Mars/Earth return trajectories it has been possible to maintain the approach velocity below 5 km/sec, the direct Earth reentry configuration has been assumed.